Two day workshop on “Recent Trends in Manufacturing”,

Dept. of Mechanical Engineering,
Amrita School of Engineering
Amrita Vishwa Vidyapeetham Coimbatore
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“CURRENT TRENDS IN AIRCRAFT STRUCTURAL REPAIR”

By

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ROUTE MAP

- INTRODUCTION
- REPAIR SEQUENCE & REPAIR PROCESS
- REPAIR CRITERIA & CAUSES OF DAMAGE
- VARIOUS REPAIR JOINTS & REPAIR SCHEMES
- REPAIR PHILOSOPHY
- REPAIR METHODOLOGY
- REPAIR CLASSIFICATION & CONCEPTS
- CAUSES & DAMAGES
- CASE STUDIES
- CONCLUDING REMARKS
- REFERENCES
- ACKNOWLEDGEMENTS

INTRODUCTION TO COMPOSITE MATERIALS

Composites materials are multi-phase materials, where the interaction of the two phases gives overall mechanical and physical properties of an efficient nature. The combination of fibrous reinforcement in a matrix is the most common form of composites.

Eg: Carbon fiber Rein forced plastic (CFRP)
Glass fiber Rein forced plastic (GFRP)

Composite = Fiber + Matrix

Fiber → Reinforcing material

Matrix → Binding Material (Hold fibers together)

Eg: Fibers

- Carbon
- Glass
- Boron
- Aramid

Eg: for Matrix

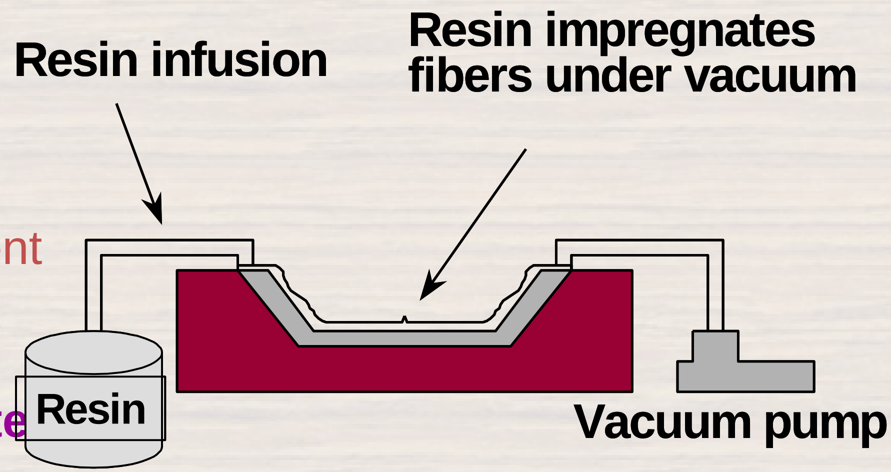
- Epoxy
- Phenolics
- Vinyl ester
- Polyester

The role of Matrix in a composite :

- It gives shape
- It makes individual fibers of the reinforcement act together
- It protects the reinforcement from the environment
- It provides the transverse strength and stiffness to laminated composites

The role of Reinforcement :

- Provide strength & stiffness
- Controls the thermal expansion co-efficient
- Provides directional properties



Advantages of Fiber Reinforced composite

- High specific strength and specific stiffness
- The multiplicity of the fibers makes a composite material highly redundant
- Before curing these materials are soft & flexible, as a result it is very easy to produce complex contoured parts
- These materials are ideally suited for repairs because the patch can be made in-situ (Cure-in-place repair).

TYPES OF COMPOSITES

There are several different types of composites used today. The most common are:

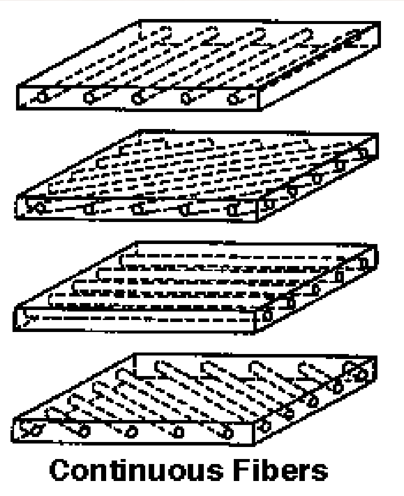
- Fiber reinforced composites
- Particulate reinforced composites

These types of composites cover a range of different material combinations.

The most common type is **polymer matrix composites**, however,

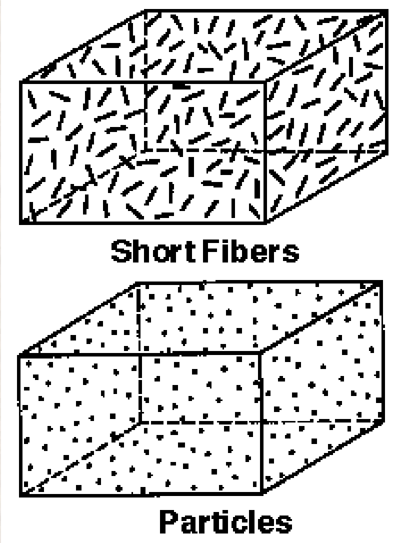
metal matrix composites, and **ceramic matrix composites** are also common,

as are **natural composites such as wood.**



Fiber reinforced composites

Carbon fiber reinforced



Particulate reinforced composites,

Concrete

WHY REPAIR?

- COMPOSITES AIRCRAFT COMPONENTS CREATE NEW CHALLENGES FOR MAINTENANCE PERSONNEL DUE TO THE UNIQUE NATURE OF THEIR DAMAGE MODES

- DEVELOPMENT OF PROPER REPAIR TECHNIQUES FOR COMPOSITES HAS BECOME VERY ESSENTIAL TO INCREASE THE CONFIDENCE OF USER IN THESE MATERIALS AND ALSO FOR ECONOMICAL REASONS.

- REPAIR OF DAMAGED COMPOSITES STRUCTURE INVOLVES MANY STEPS AND ALSO QUITE INVOLVED EXERCISE

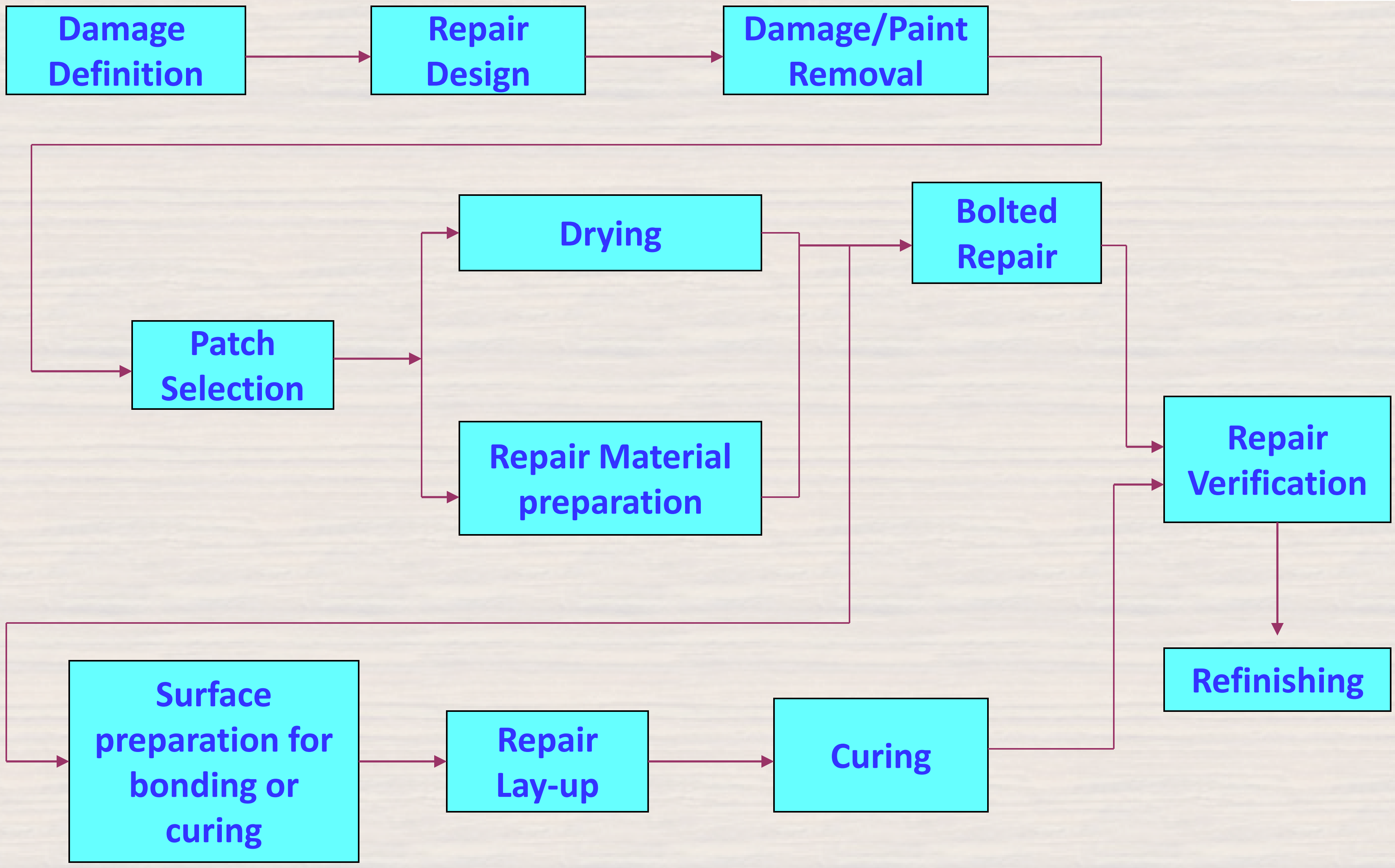
- IN ORDER TO FACILITATE THE REPAIR TEAM AND SAVE THE TIME IT IS NEEDED TO DEVELOP SPEEDY FIELD REPAIR SCHEMES.

REPAIR SEQUENCE

The repair sequence routine:

- Find the damage
- Define extent of damage
- Calculate damage effects and
    - a) Use as is
    - b) Design a repair
    - c) Scrap the part
- Specify the repair method, if repair is the option
- Accomplish the repair
- Evaluate the repaired part.

REPAIR PROCESS



HIGHLIGHTS TYPICAL DEFECTS / DAMAGES AND POSSIBLE CAUSES

DEFECTS/DAMAGES	CAUSES
MANUFACTURING	
Porosity	Defective tooling/lack of uniform pressure
Voids	Poor Process Control
Delamination	Poor process control/ presence of foreign material like release film/ backing film/ peel ply etc. Improper machining operations.
Disbonds (in bonded structures)	Poor process control/ improper tooling/ lack of perfect matching in mating surfaces/ improper surface preparation
Surface damage	Poor process control/ in adequate release preparation etc., bad handling
Misdrilled holes	Poor process/ faulty jigging

SERVICE	
Cuts / Scratches	Mishandling
Abrasion / erosion	Rain / Grit
Burning	Engine exhaust gasses
Delaminations	Impact damage
Disbonds	Improper design/ over load
Hole elongation	Over load/ improper design/ bearing failure
Dents	Mishandling / improper edge protection
Penetration	Battle Damage etc.
ENVIRONMENTAL	
Surface oxidation Delamination/ Disbonds	Lightening strike/ over heat poor sealing (moisture expansion) thermal spike steam formation
Core Corrosion Surface Swelling	Moisture penetration use of undesirable solvents in paints etc

REPAIR PROCEDURES

PROCEDURE	APPLICATION
NON-PATCH REPAIR PROCEDURE FOR MINOR DAMAGE	
Resin Injection	Connected voids Small delaminations Small disbonds
Potting or Filling	Minor depressions Skin damage in honeycomb panels Core replacement in honeycomb panels Fasteners hole elongation
Surface coating	Seal honeycomb panels Restore surface protection
PATCH REPAIRS FOR MAJOR DMAGE	
Bonded external patch graphite/epoxy (Co-cured; pre-cured layers ;)	Repairs to skins, particularly on honeycomb panels, upto 16 plies thick. Well suited for field application
Bonded scarf patch graphite/epoxy (usually/co-cured)	Repairs to skins 16 to 100 plies thick, holes upto 100mm. Size may be difficult to employ under field conditions,
Bolted external patch titanium alloy (usual) aluminum alloy	Repairs to monolithic skins 50-100 plies upto 100mm. Suited for field applications.

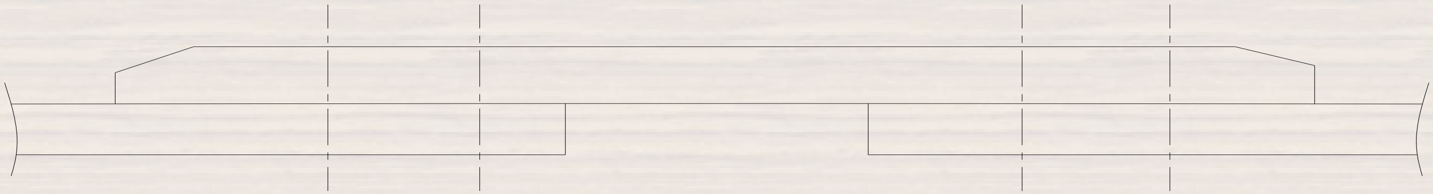
PATCH REPAIRS

Patch repairs are generally employed to repair major damage and essentially involve replacing the lost load patch with new material joined to the parent structure.

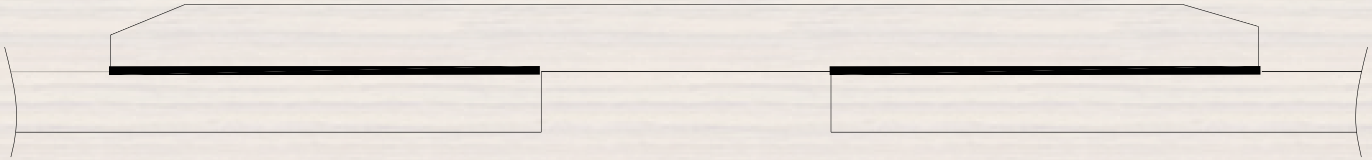
Patch repairs are widely classified into 3 parts Viz.,

1. Bonded patch repair.
2. Bolted patch repairs.
3. Hybrid Repairs. (Mixed Joints)

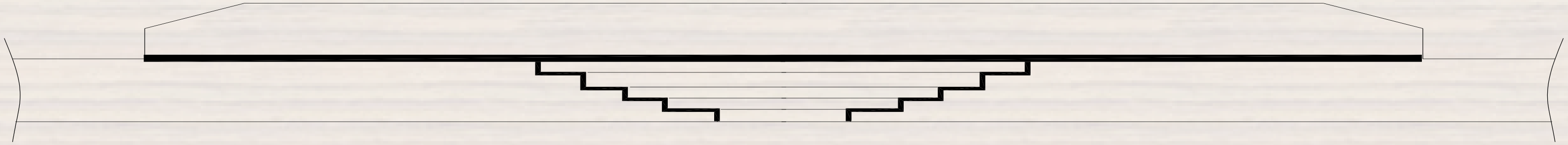
THE MOST COMMON METHODS OF COMPOSITE REPAIR



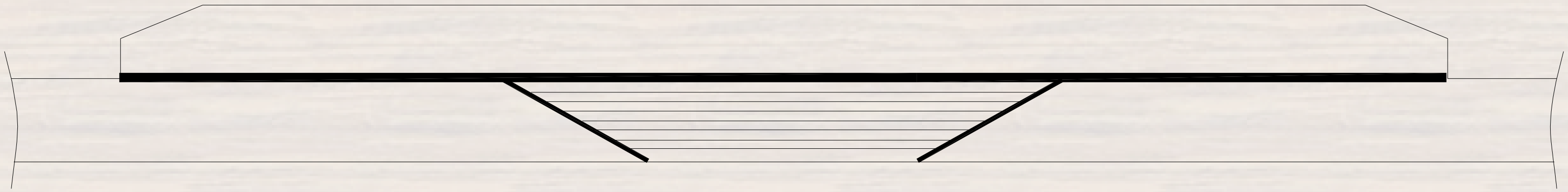
(a) Bolted Repair



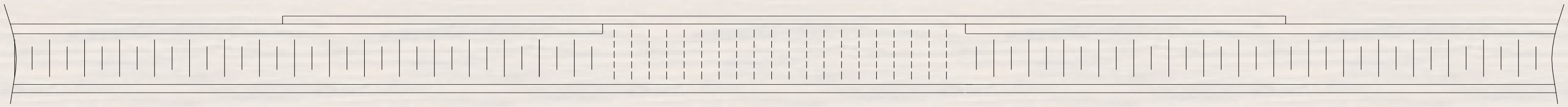
(b) Bonded Repair



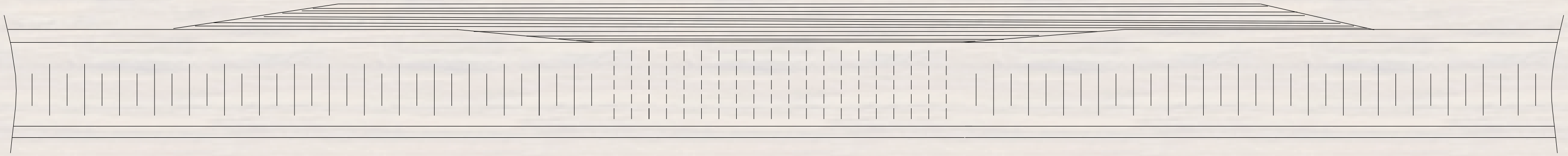
(c) Stepped-lap Repair



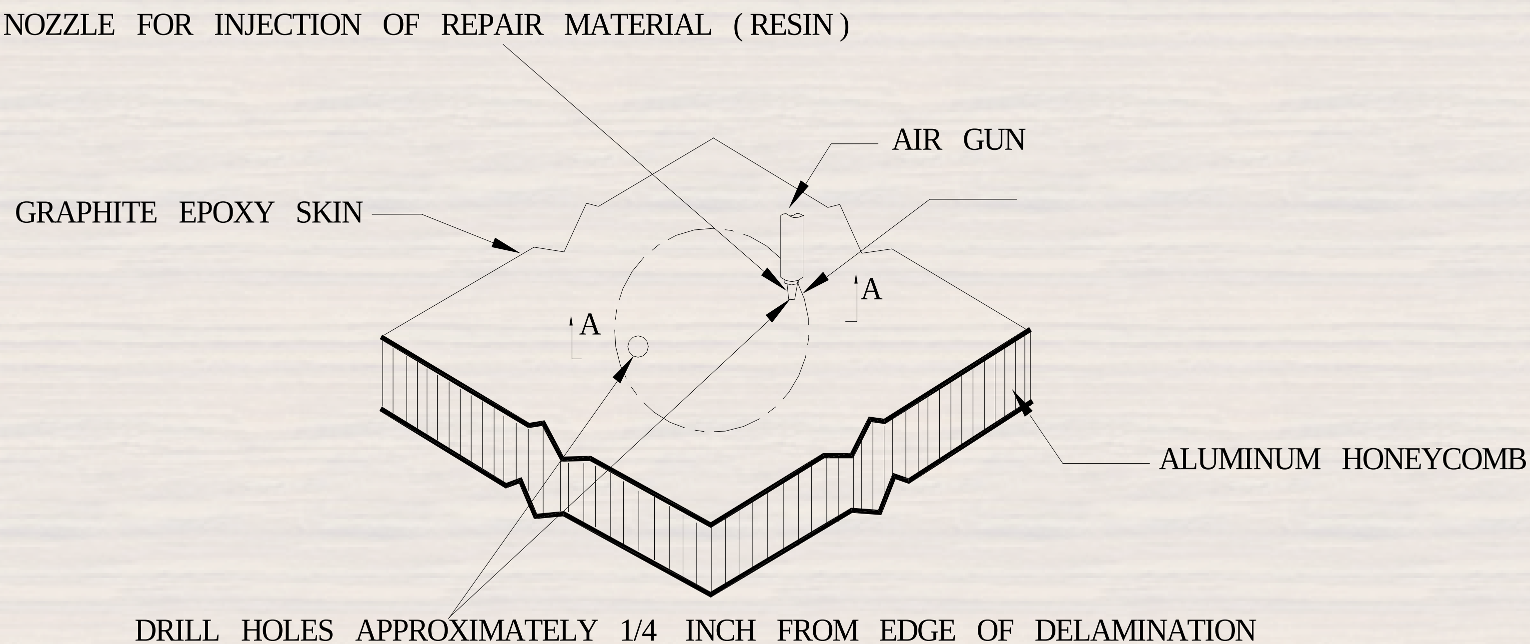
(d) Scarf Repair

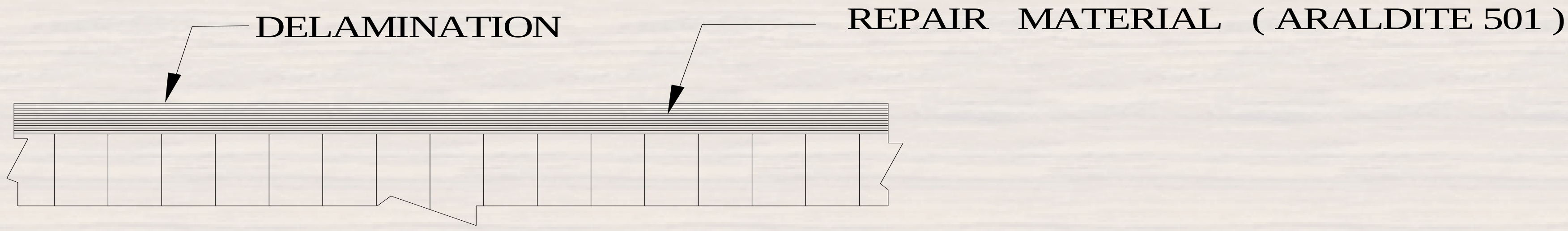


(e) Honeycomb Repair



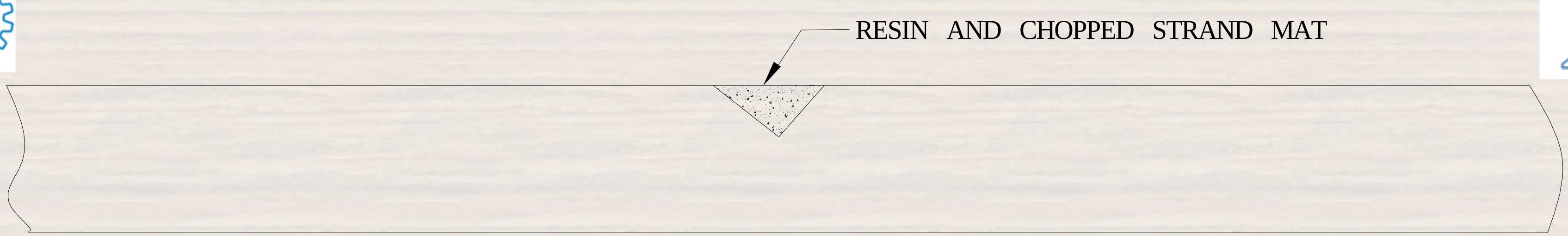
(f) Alternative honeycomb Repair



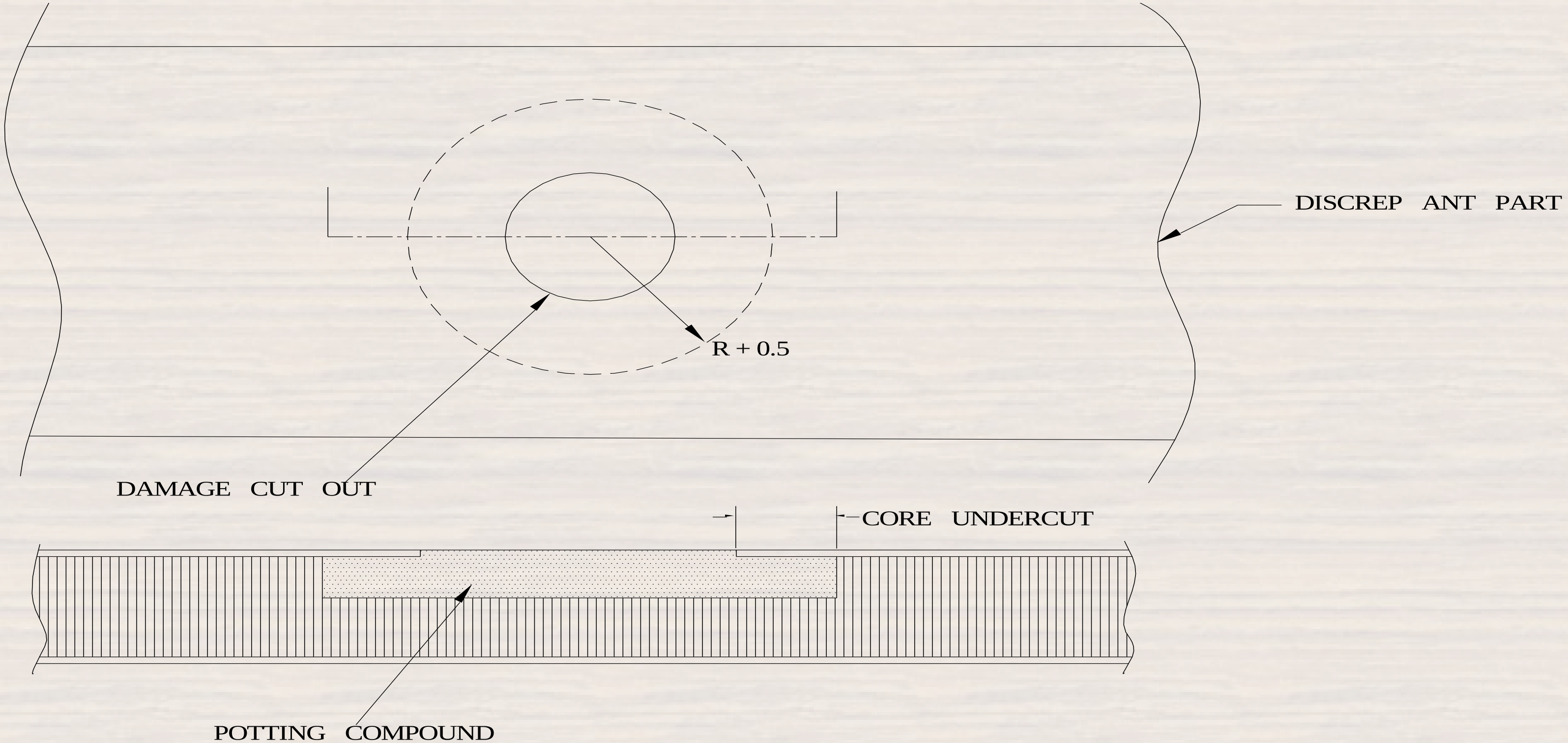


SECTION A-A

REPAIR OF DELAMINATION ON HONEYCOMB PANEL BY RESIN INJECTION



REPAIR V - GROOVES BY POTTING

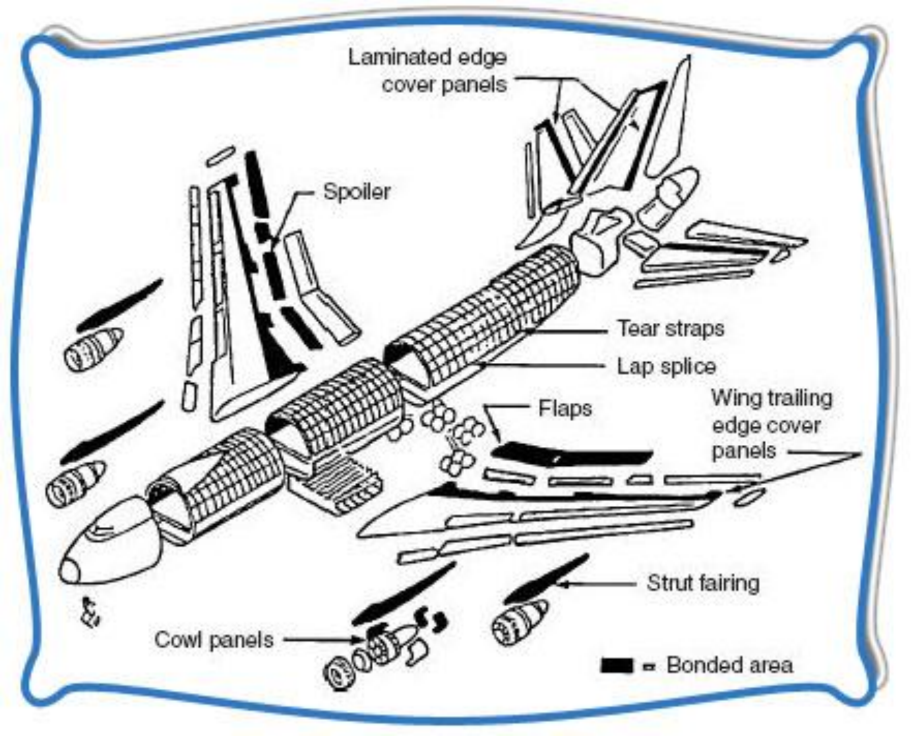


POTTED REPAIR TO A DAMAGED HONEY COMB REGION

STRUCTURAL JOINTS

- INTRODUCTION
- MECHANICALLY FASTENED JOINTS
- SNAP JOINTS
- BONDED JOINTS
- HYBRID JOINTS
- SUMMARY

COMPOSITE BONDED – BOLTED JOINT OF FLAP TRACK BEAM OF A-400M



COMPOSITE BONDED JOINTS IN BOEING-777

INTRODUCTION

- An airplane is a large assembled structure with large number of sub-assemblies in it.
- The structures are assembled by bonding, fastening, welding etc.,
- The joints are major source of failure in an assembled structure
- The assembly joints are a major source of stress concentration
- So it is important to ensure that the joints either bonded or fastened should not reduce the efficiency of the structure

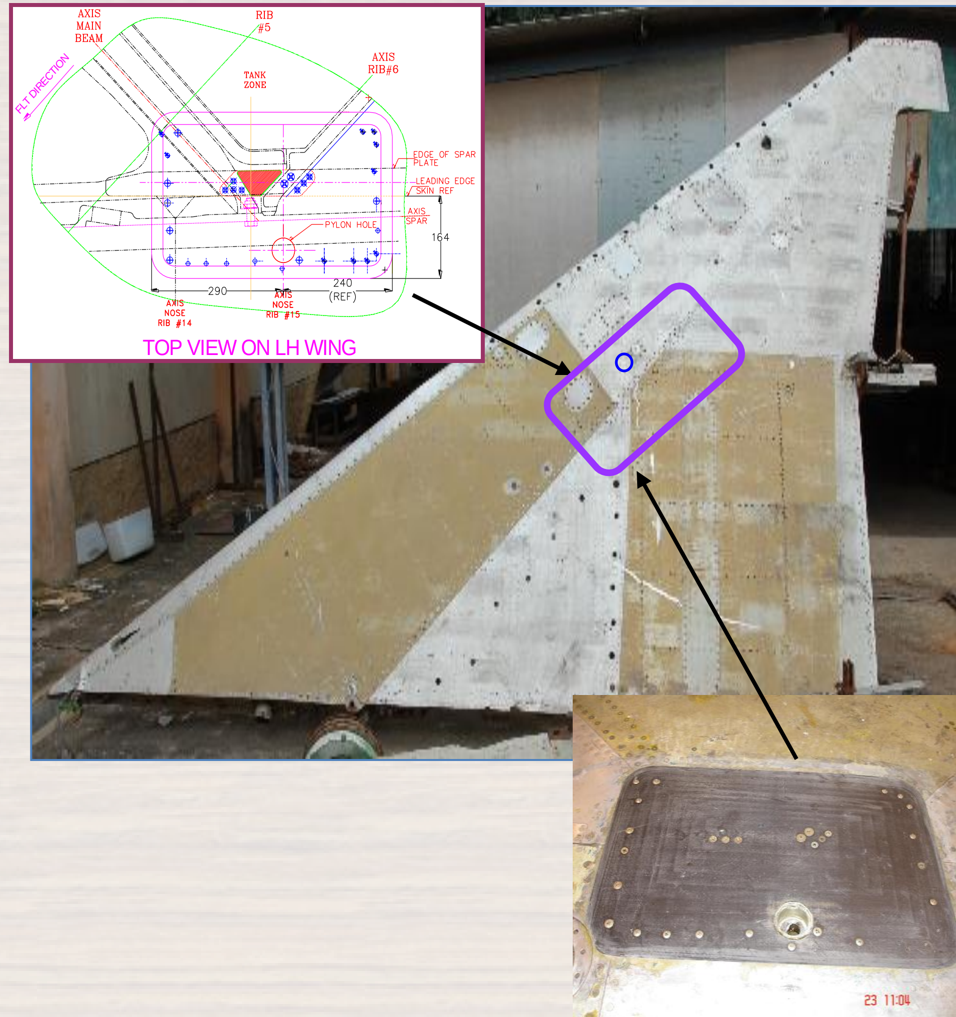
COMPARISON OF JOINING METHODS

Method		Anticipated Benefits	Limitations
Mechanical fastening		• Mature Technology • Baseline for cost data • Could supplement weld/bond assembly methods	• Low risk • Increased weight • Labor Intensive • Requires secondary seal • Shimming fit-up stress
Adhesive bonding		• Reduced fastener count/weight	• Moderate risk • Cure cycle required • Tooling
Thermoplastic Welding	• Resistance	• Can be automated process • Continuous weld • Reduced fastener count/weight	• Moderate risk • Requires 2 side access
	• Ultrasonic	• Can be automated process • Possible continuous weld • Reduced fastener count/weight	• Moderate risk • Requires 2 side access
	• Induction	• Requires 1 side access • Can be automated process • Continuous weld • Reduced fastener count/weight	• Moderate — high risk • Requires magnetic susceptor mat'l
Cocuring		• Total homogeneous weld joint • Probable elimination of seal	• Low risk • Part size/shape limited

DESIGN OF A COMPOSITE JOINT

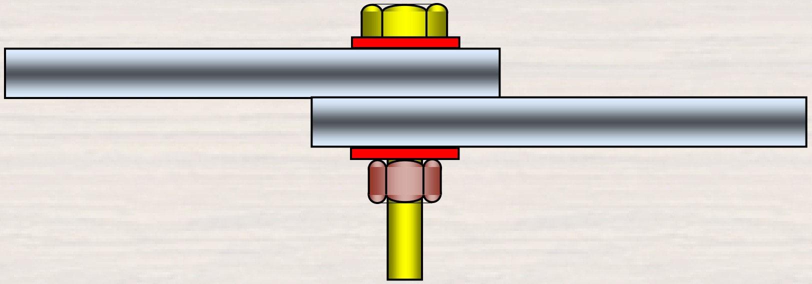
Six basic factors in the design of a composite joints:

- The loads to be transferred
- The region
- The geometry of the structures
- Service environment
- Weight & cost efficiency
- Reliability of joint

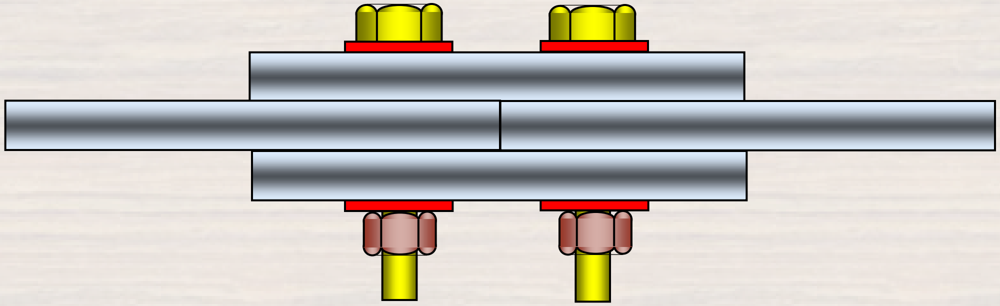


MECHANICALLY FASTENED JOINTS

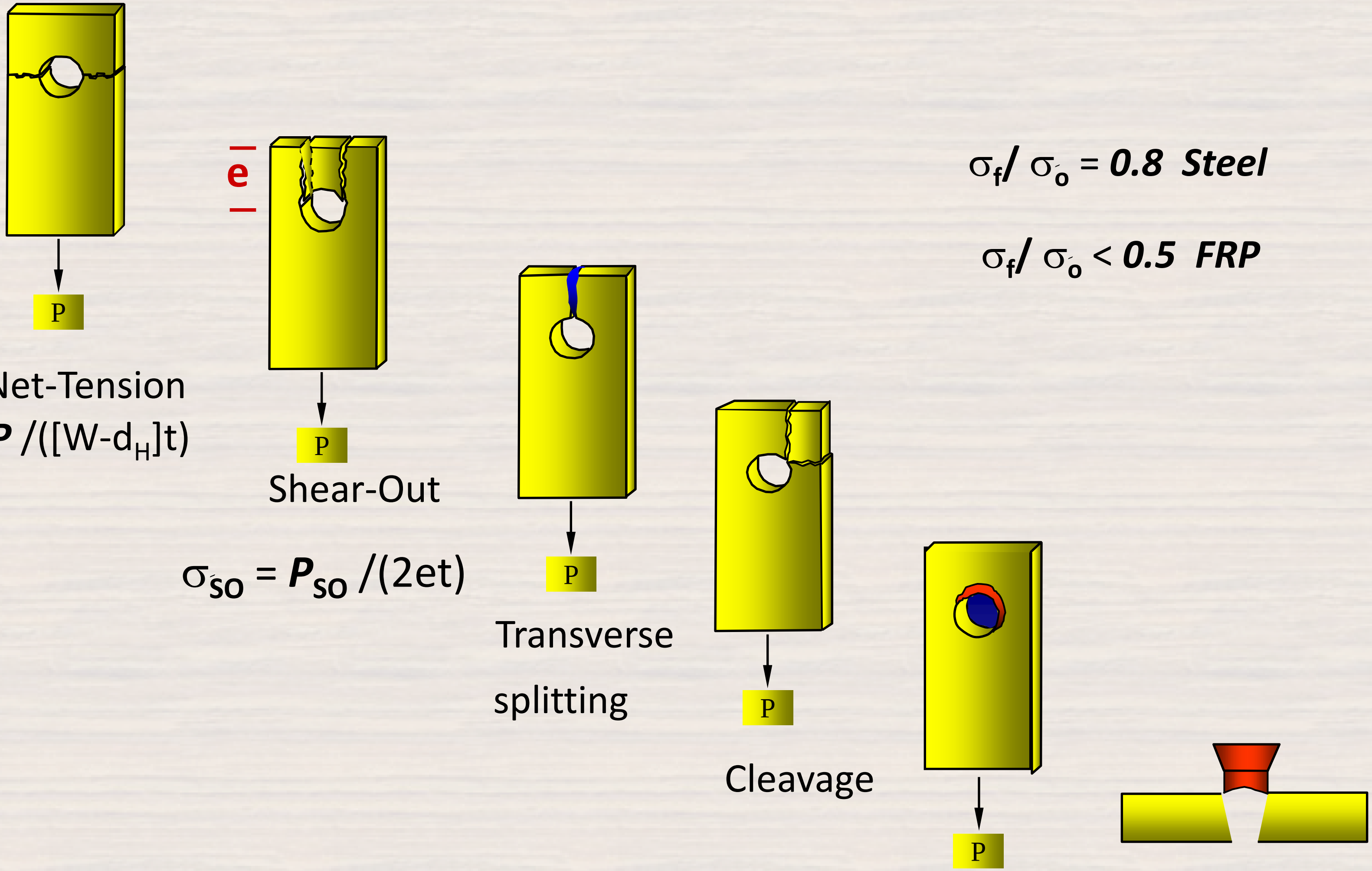
Single lap bolted joint



Double Strap bolted joint



FAILURE MODES OF BOLTED JOINTS



σ_f / σ_o = ***0.8 Steel***

σ_f / σ_o < ***0.5 FRP***

Net-Tension

$\sigma_N = \boldsymbol{P} / ([W - d_H] t)$

Shear-Out

$\sigma_{SO} = \boldsymbol{P}_{SO} / (2et)$

Transverse

splitting

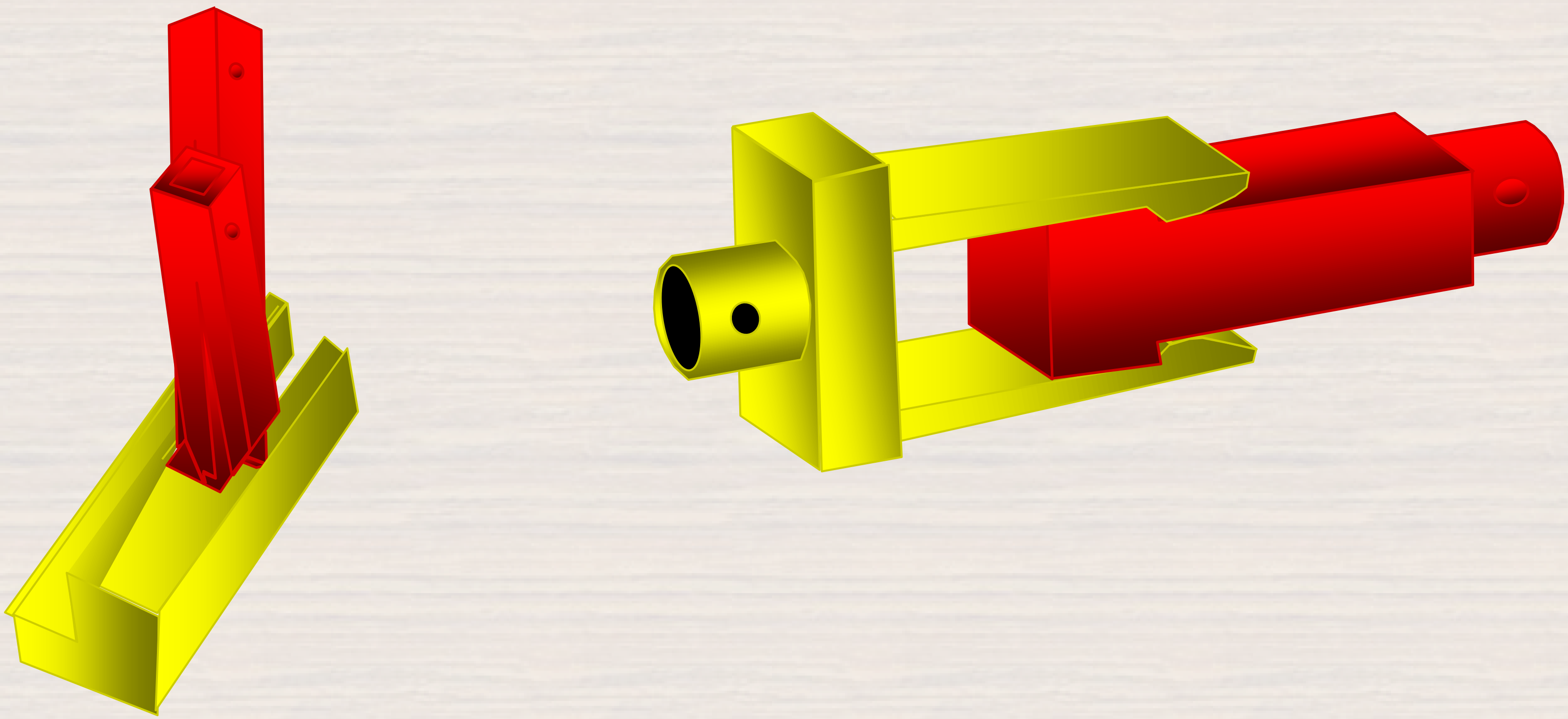
Cleavage

Bearing

Pull-out

$\sigma_b = \boldsymbol{P} / (d_B t)$

SNAP JOINTS

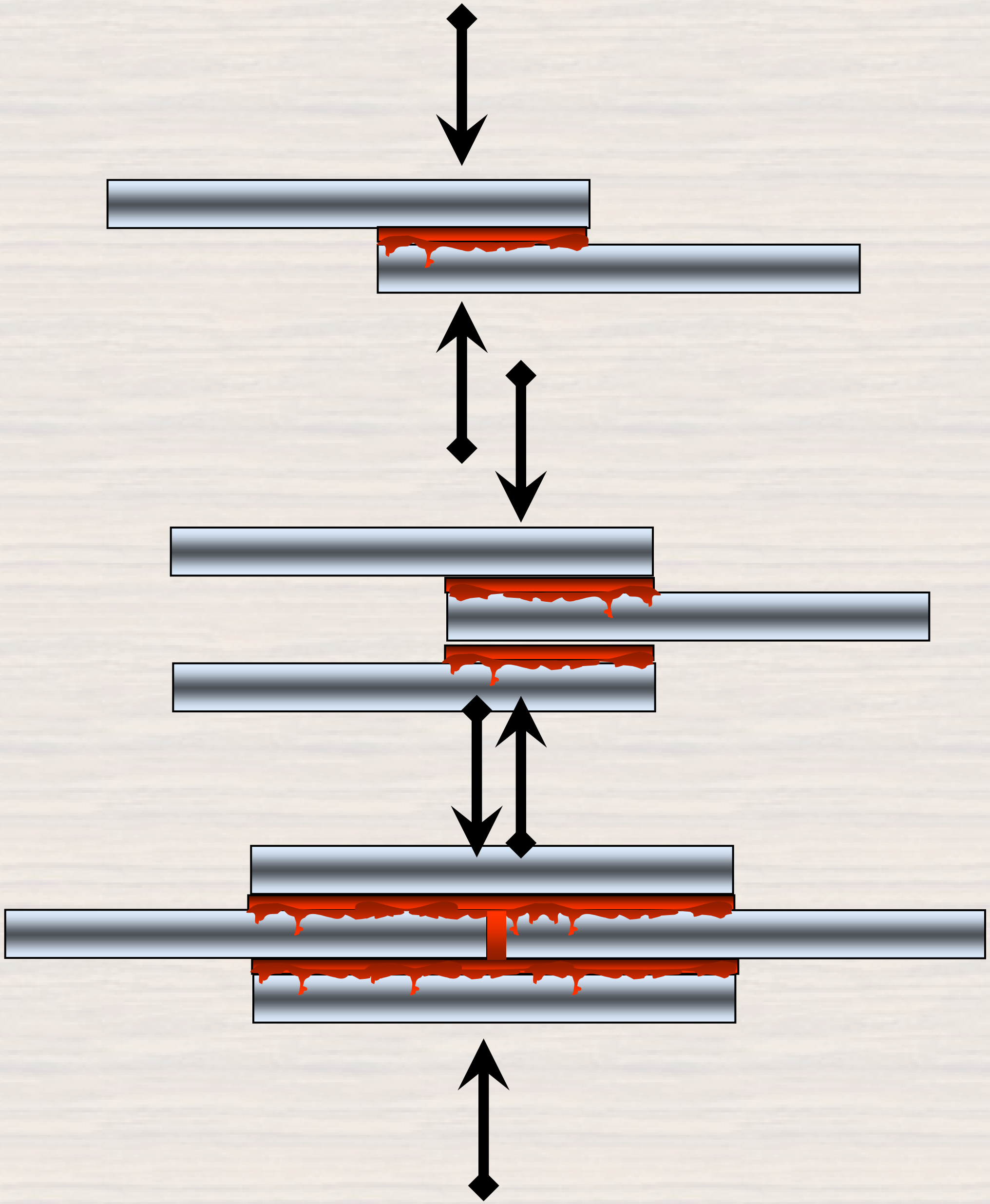


SNAP JOINTS

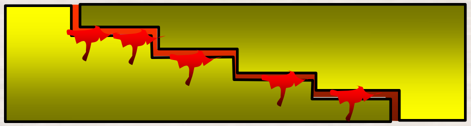
(a) Composite Transmission Tower *(b) Truss Joint*

BONDED JOINTS

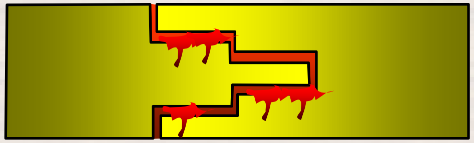
- Single lap
- Double lap
- Double strap



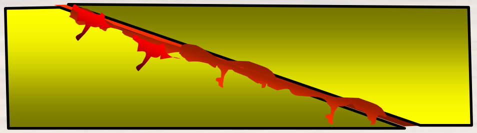
BONDED JOINTS



Stepped lap



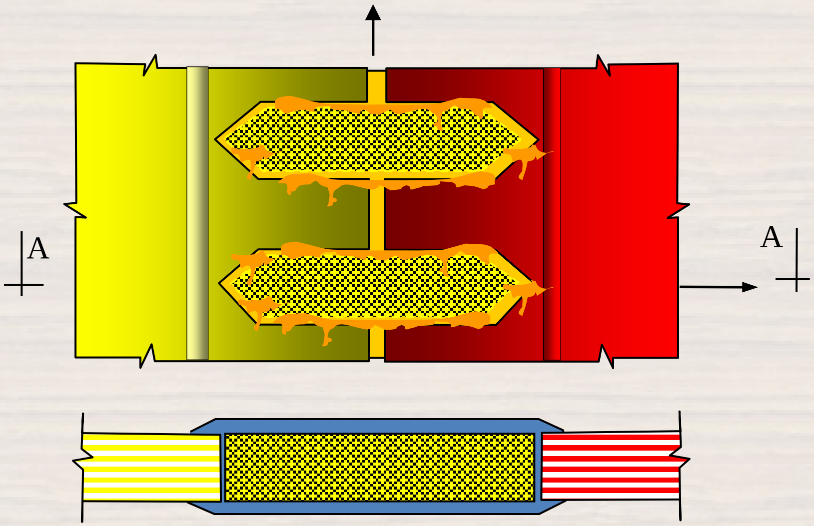
Double stepped lap



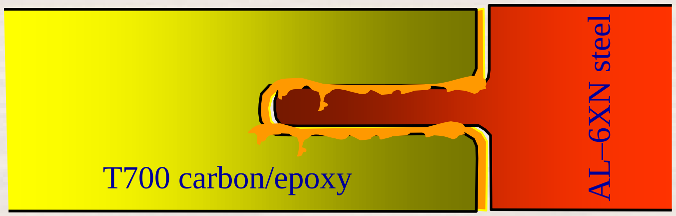
Single scarf

Double scarf

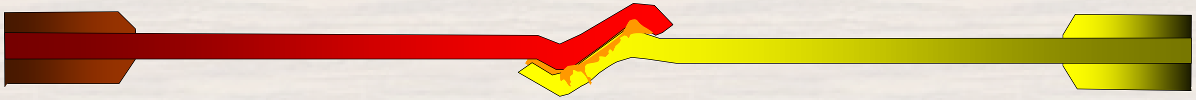
BONDED JOINTS



New design with flat, pointed inserts

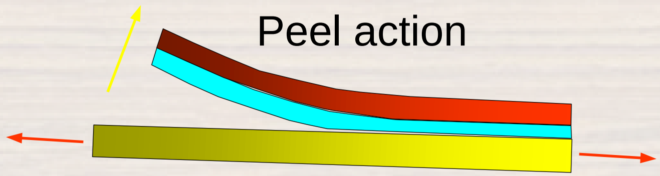


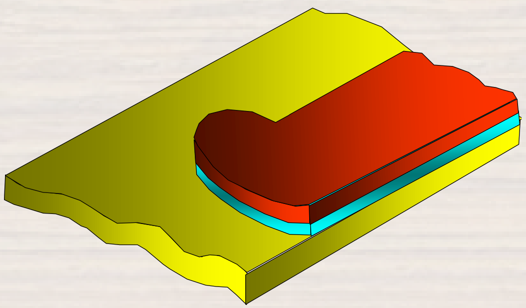
The general geometric parameters for a tongue-and-groove joint.



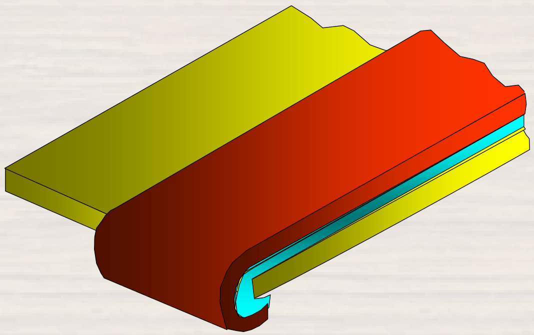
Wavy-lap joint main dimensions

PREVENTION OF PEELING FAILURE IN ADHESIVE JOINTS

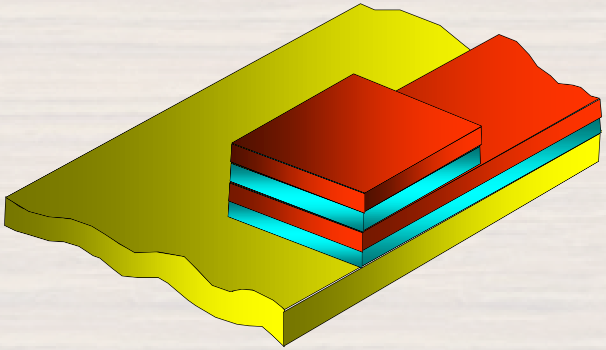




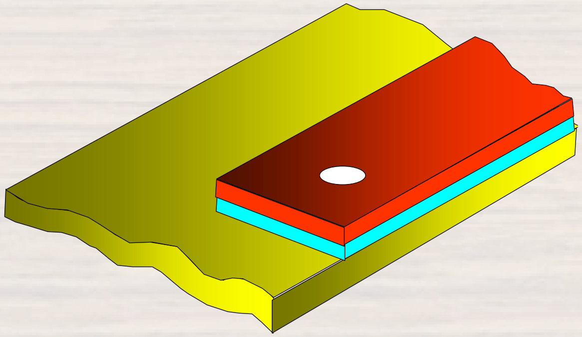
Increase area



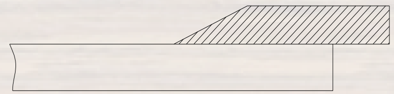
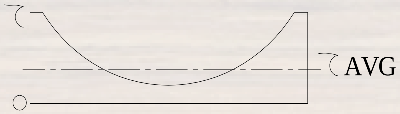
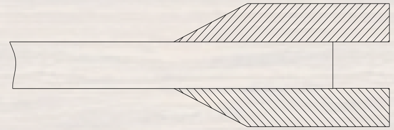
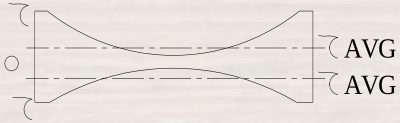
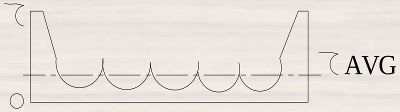
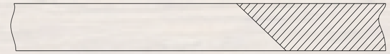
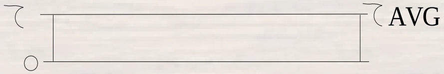
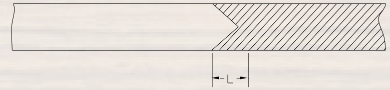
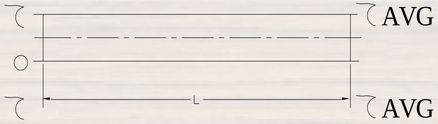
Bead end (if possible)



Increase stiffness



Rivet, bolt or spot weld

TYPE	SCHEMATIC	SHEAR STRESS
SINGLE OVERLAP		
DOUBLE OVERLAP		
SINGLE STEP - UP		
SINGLE SCARF		
DOUBLE SCARF		

MAIN TYPES OF JOINT CONFIGURATIONS EMPLOYED FOR BONDED PATCH REPAIRS AND THE RESULTING SHEAR STRESS DISTRIBUTIONS

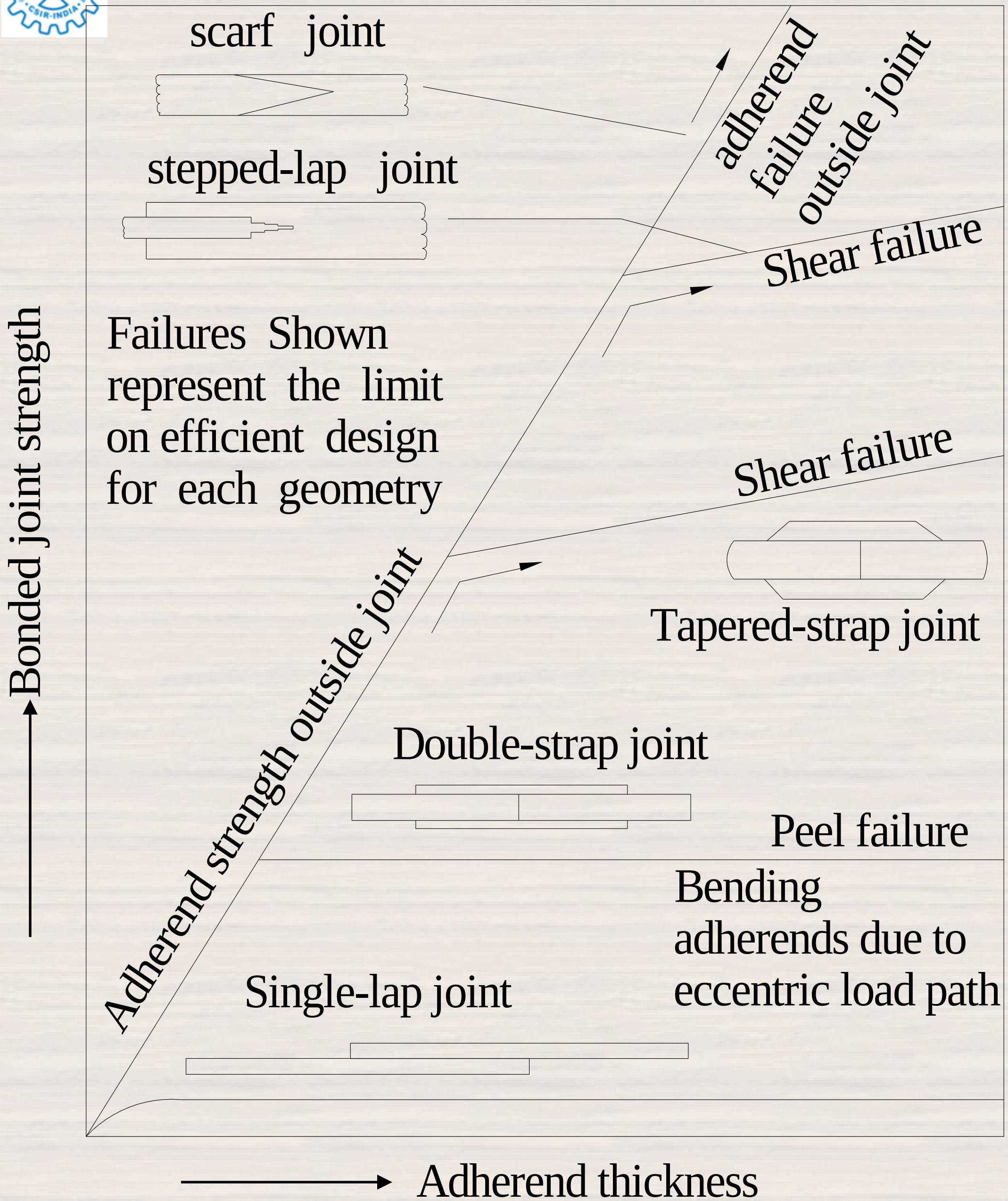
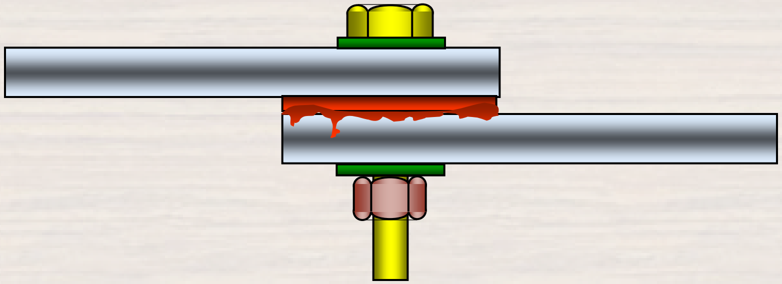
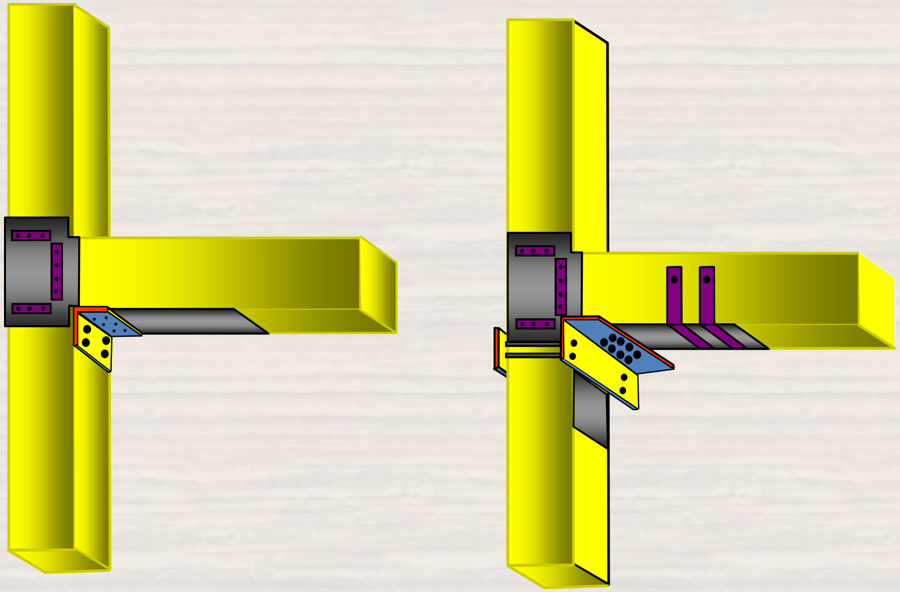


Fig. Effect of adherend thickness on relative strength of different joint types

HYBRID JOINTS





Retrofitting schemes

COMPARISON OF BONDED, BOLTED & BONDED - FASTENED JOINTS FOR COMPOSITES

REPAIR METHOD	PROS	CONS
Adhesive Repair (Bonded Repair)	➢Lighter weight ➢Distribution of load over wider area ➢Better for thin laminates	➢Degradation in service due to temperature and humidity ➢Requires surface preparation ➢Difficult to inspect ➢More difficult to perform correctly
Mechanically Fastened repair (Bolted Repair)	➢Not adversely affected by temperature and humidity ➢Less surface preparation required ➢Easy to repair and suitable for field repair ➢Generally visual inspection required, and sometimes may need Eddy-current to inspect holes	➢Adds weight and bulk ➢Requires holes in weakened members ➢Produces stress concentration ➢Clearance between bolt and hole distributes load unevenly ➢Susceptible to delamination damage from drilling/machining
Bonded and Fastened (Hybrid Repair)	Useful if peel loads cannot be reduced in levels acceptable for solely bonded joints	• Cannot be easily disassembled • Increased weight • Bonding and fastening operation may increase the production costs.

SUMMARY

- Hybrid joints give better static as well as fatigue performance than any conventional joint configurations, i.e., bonded or hybrid joints.
- To provide a fail-safe design, use hybrid joints(bonded/bolted) in composites structures

Development of Improved Hybrid Joints for Composite Structures

C.T. Sun, Bhawesh Kumar, School of Aeronautics and Astronautics,

and

P. Wang, R. Sterkenburg, Aviation Technology,

(Purdue University, West Lafayette.Indiana 47907)

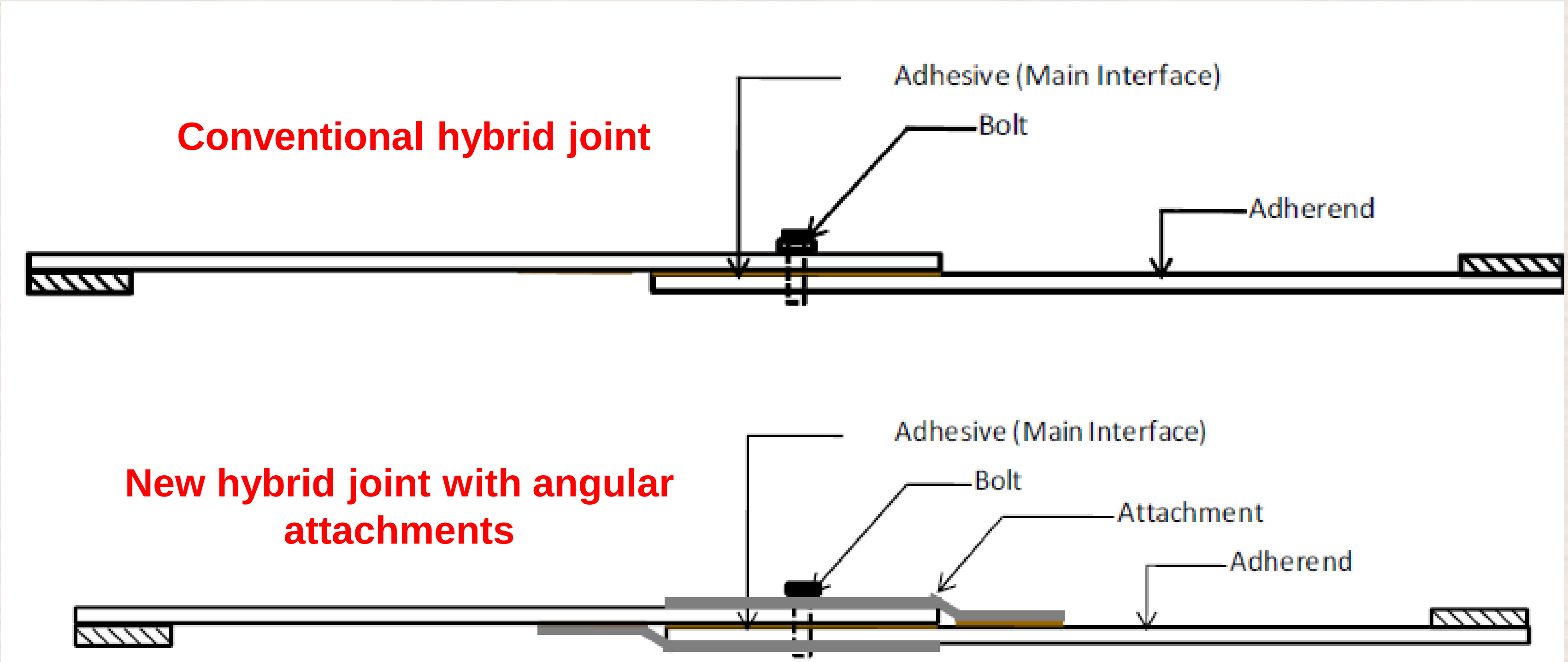
ASTM STP 1455 (2004)

ABSTRACT

- In the designs of hybrid joints, the adhesive bond supports the entire load of the joint as long as the adhesive is intact and bolts do not have any contribution in load transfer before the failure of bonded surfaces.
- Therefore, the hybrid joint design has the same strength as the bonded joint.
- A new hybrid joint design was proposed for composite lap joints.
- It use a small flat piece of composite laminate attachment to create an alternate load path to transfer part of the load from the adherend to the bolt
- Experimental investigations revealed that the strength of the new hybrid joint was significantly greater than that of the conventional hybrid joint design.
- Two-dimensional finite element analyses were performed to provide the explanation of this strength enhancement.

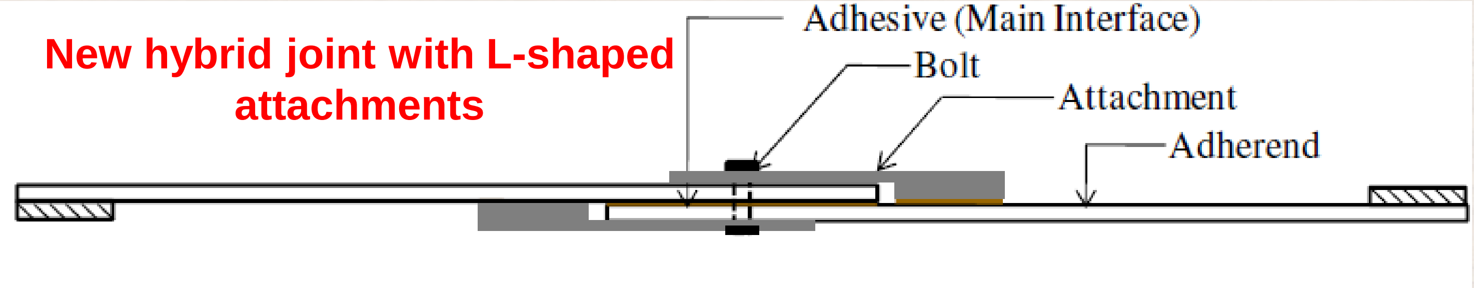
INTRODUCTION

- The literature survey reveals the failure behind hybrid joints are due to absence of bolts in load taking process.
- Therefore experiments were conducted to design a new hybrid joint
- Investigators proposed an idea of new hybrid joint using attachments to provide additional load transfer paths at joints.
- The increase in joint strength of the new hybrid joint with attachments was found to be approximately 80% higher as compared to the conventional hybrid joint



EXPERIMENTS

- Based on literature survey author proposed a new hybrid joint design for composites
- Use of straight composite laminate attachments instead of bent attachments.
- In the proposed hybrid joint design, the bolt contributes to load bearing as soon as the joint is loaded.



- Experiments were conducted on five different joint configurations, bonded joint, bolted joint, conventional hybrid joint and hybrid joint with attachments.
- The AS4/3501-6 carbon/epoxy Prepreg tape was used to fabricate a 20-plied laminate

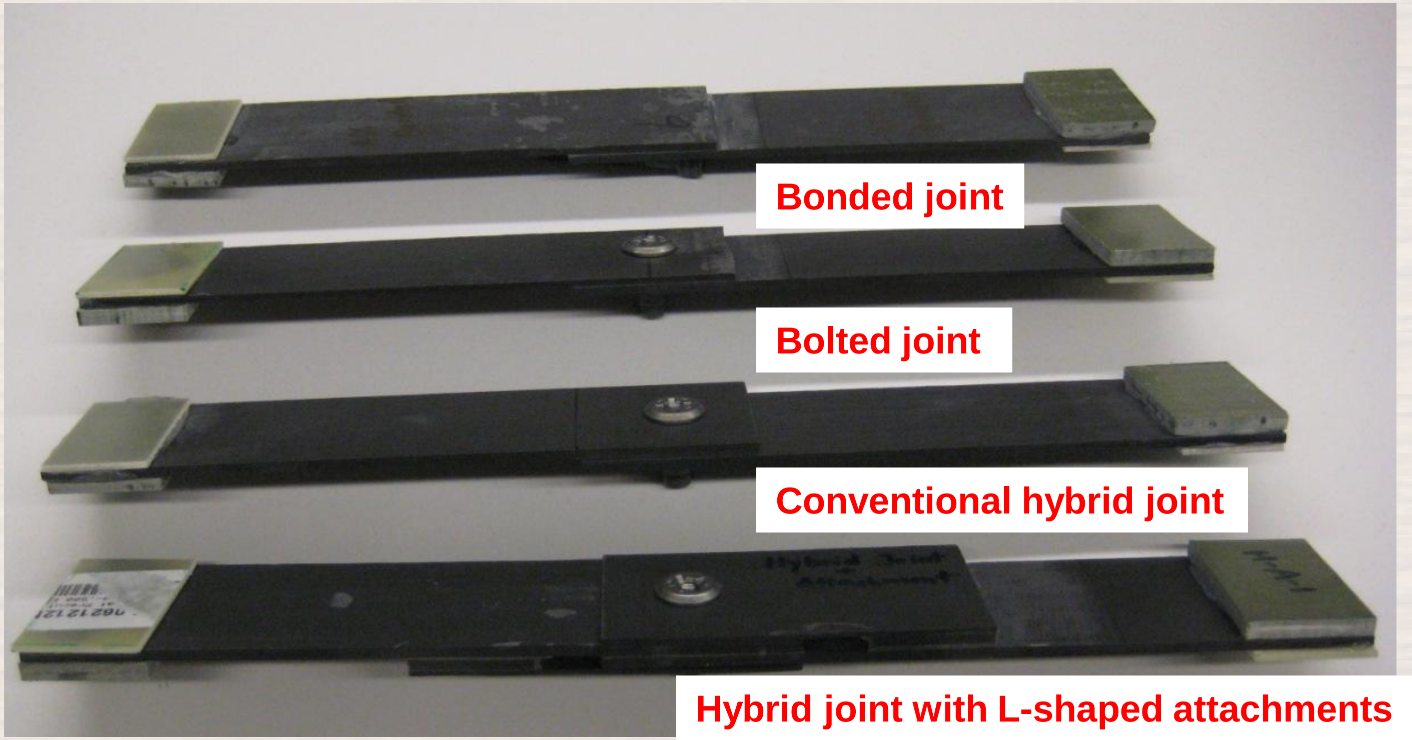
EXPERIMENTS

Attachment preparation:

- L-SHAPED ATTACHMENT
  - ✓ L-shaped attachments were composed of two flat pieces of composite laminates.
  - ✓ The long flat piece was prepared with the same Prepreg material with a stacking
  - ✓ Sequence half of the total thickness of the adherend.
  - ✓ The smaller piece was cut from the same panel for the adherends
  - ✓ These two flat pieces of attachments were bonded together using structural paste adhesive Hysol EA 9394.
- STEPPED ATTACHMENT
  - ✓ Four 0-deg AS4/3501-6 plies were stacked together and cured in the autoclave.
  - ✓ Rectangular pieces of dimension 25mm x 38mm were cut using water jet cutting machine.
  - ✓ Four of these small rectangular pieces wereglued together in a stepped fashion.
  - ✓ The top part of this stepped attachment was attached using Hysol to the top flat attachment as in the L-shaped attachment.

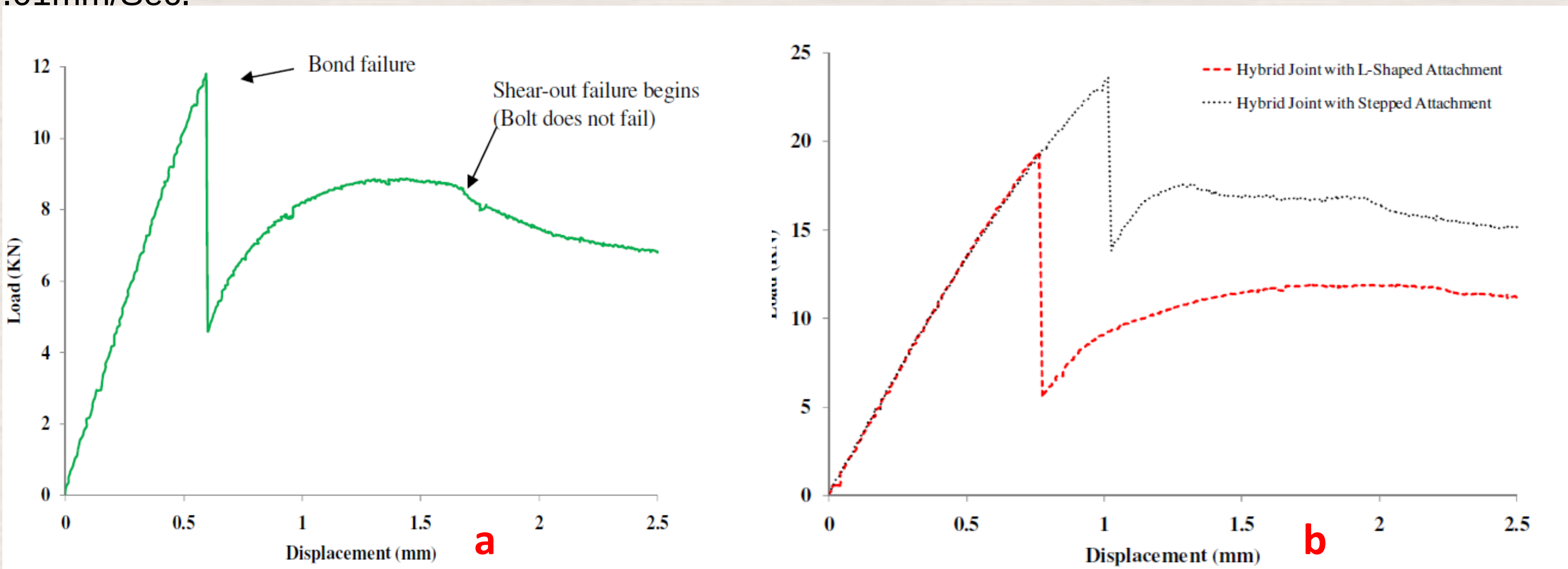
EXPERIMENTS

- A diamond drill bit was used to drill a hole at the center of the overlapped region of the joint
- A titanium bolt was placed in place and fastened.
- Tabs were attached to the adherends to mainly minimize load eccentricity in the single lap joint.

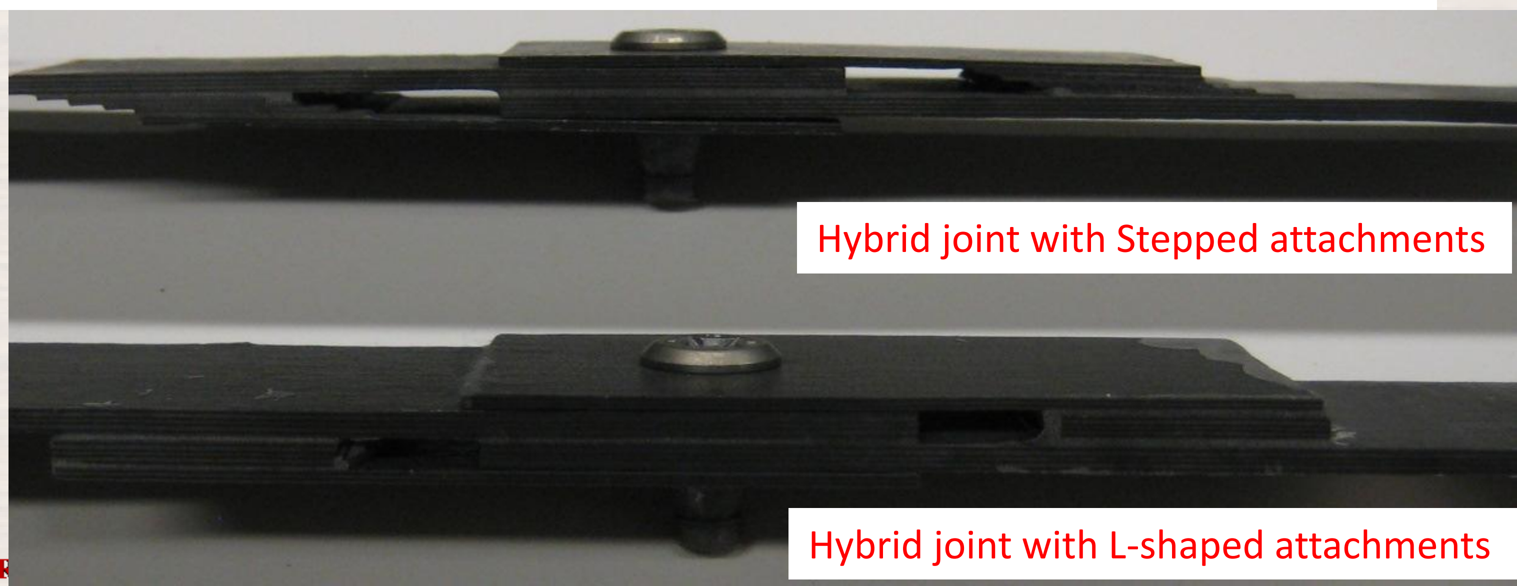


EXPERIMENTS

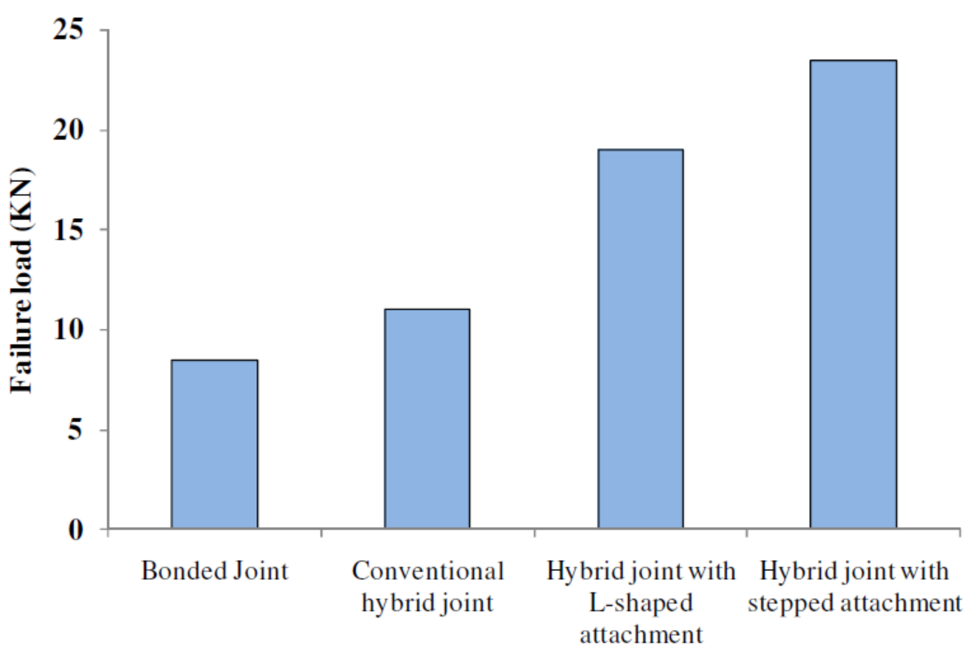
- The experiments were conducted on an MTS 22Kips machine at a crosshead displacement rate of 0.01mm/Sec.

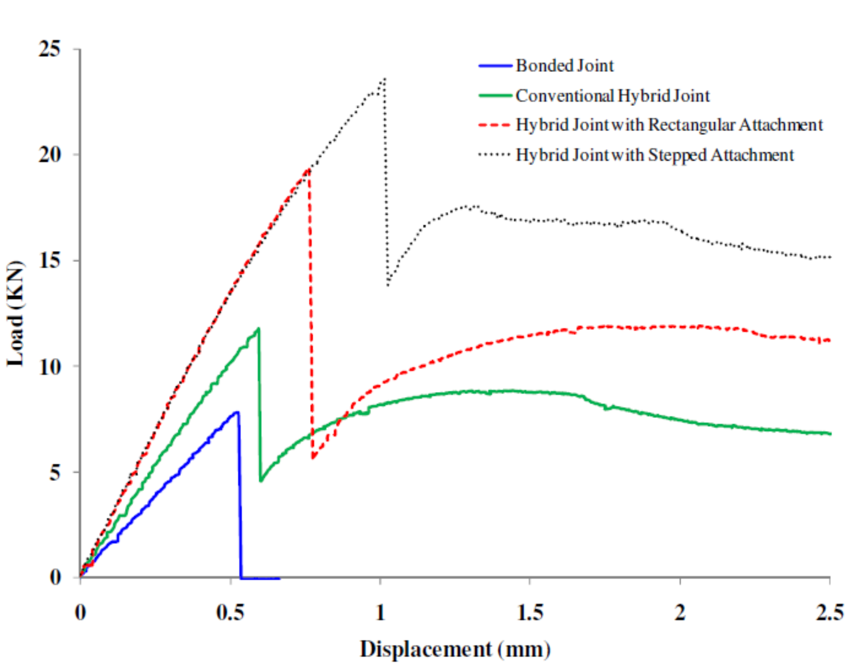


Load-displacement curve for the
a) conventional hybrid joint
b) hybrid joint with attachments



EXPERIMENTAL RESULTS





Comparison of failure loads for four joint configurations

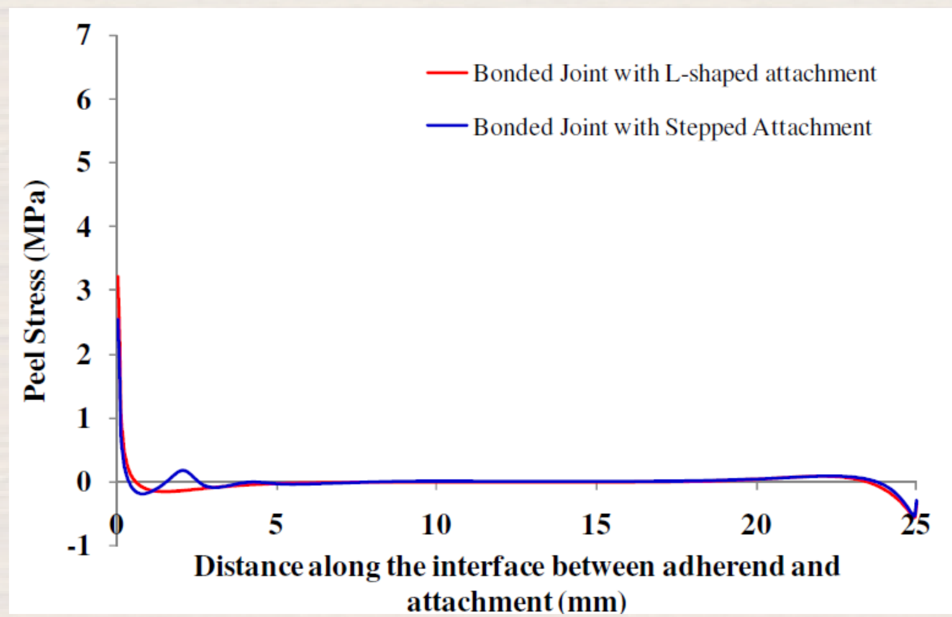
Load-displacement curves for all joints

The hybrid joint with stepped attachments yields the best strength among all these joint configurations. The failure load for the hybrid joint with stepped attachment is 160%, 120% and 20% higher than the bonded joint, the conventional hybrid joint and the new hybrid joint with L-shaped attachments, respectively

FINITE ELEMENT ANALYSIS

- Plane strain finite element analysis was performed using commercial code Abaqus 6.8-1[13] to find the effect of attachments on the hybrid joint
- Three cases are analyzed, bonded joint, hybrid joints with L-shaped attachments and stepped attachments

A typical stress pattern for three joint configurations



Peel stress distribution along the interface of the attachment and the adherend

SUMMARY

- Hybrid joints give better static as well as fatigue performance than any conventional joint configurations, i.e., bonded or hybrid joints.
- It was found that 22% of the applied load was transferred to the attachment and the remaining 78% was transferred to the other adherend through the main interface
- Hybrid joints with stepped attachments performed better than hybrid joints with L-shaped attachments.
- The joint strength were 75% and 115% for the L-shaped and stepped attachments, respectively, over the conventional hybrid joint.
- Finite element analysis result indicates the hybrid joint with stepped attachments experiences the lowest peel stress among all the joint configurations considered

CASE STUDIES

1. BONDED REPAIRS
   - MIG-21 RUDDER
   - SEA HARRIER FUEL TANK
2. RIVETED REPAIR
   - LCA FIN
   - LCA WING SPAR
3. HYBRID REPAIR
   - MiG-23 MAIN LANDING GEAR BEAM

DEVELOPMENT OF COMPOSITE REPAIR ON MiG-21 METALLIC RUDDER

COMPOSITE REPAIR ON METALLIC RUDDER

COMPONENT : MiG-21 RUDDER

- **TYPE OF DAMAGE :** Crack in the skin
- **CAUSE OF DAMAGE :** F.O.D
- **REPAIR REQUIREMENTS :**

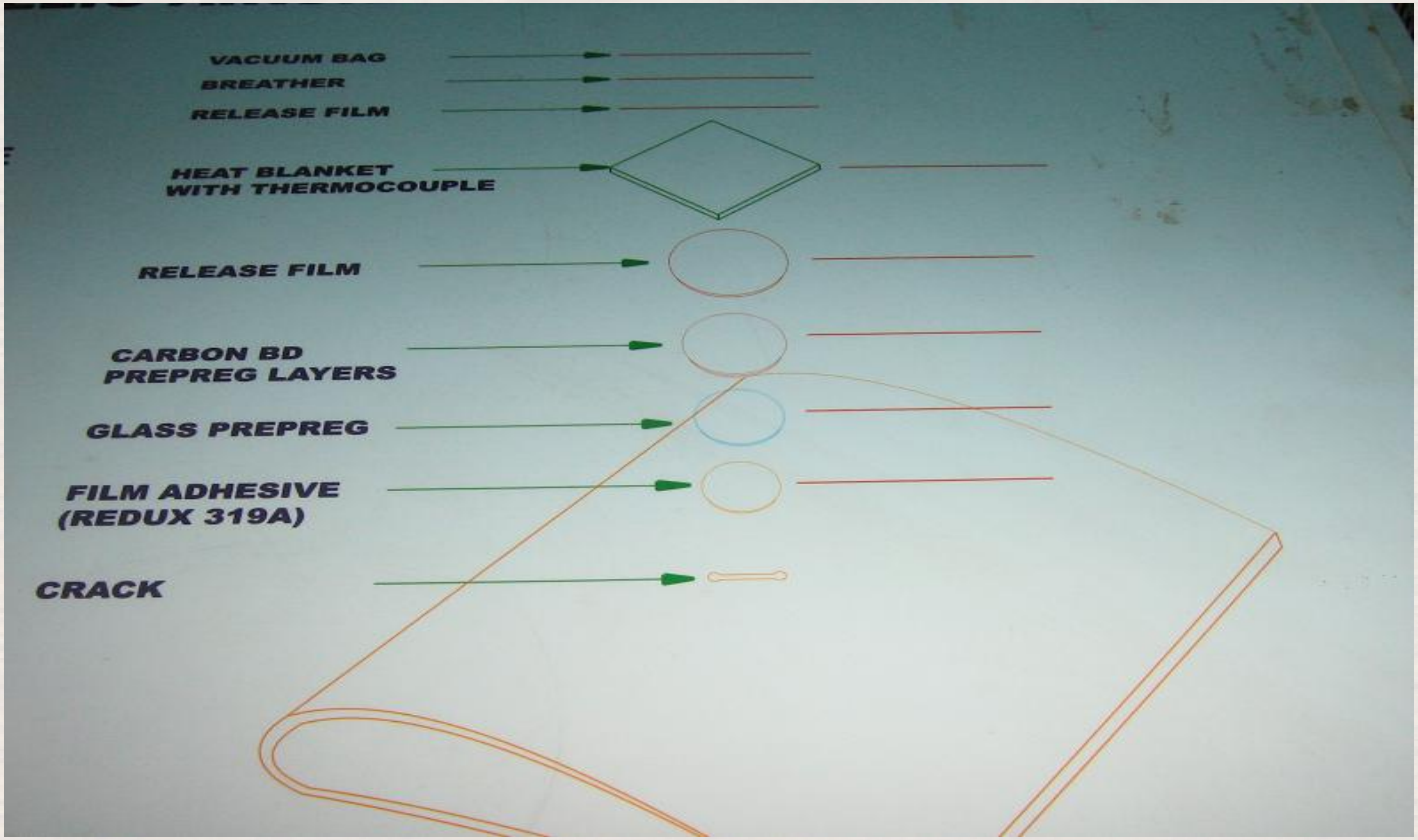
  Restoration of structural integrity

- **REPAIR SCHEME:**

  External patch (cure in place repair
  Using cure controller)

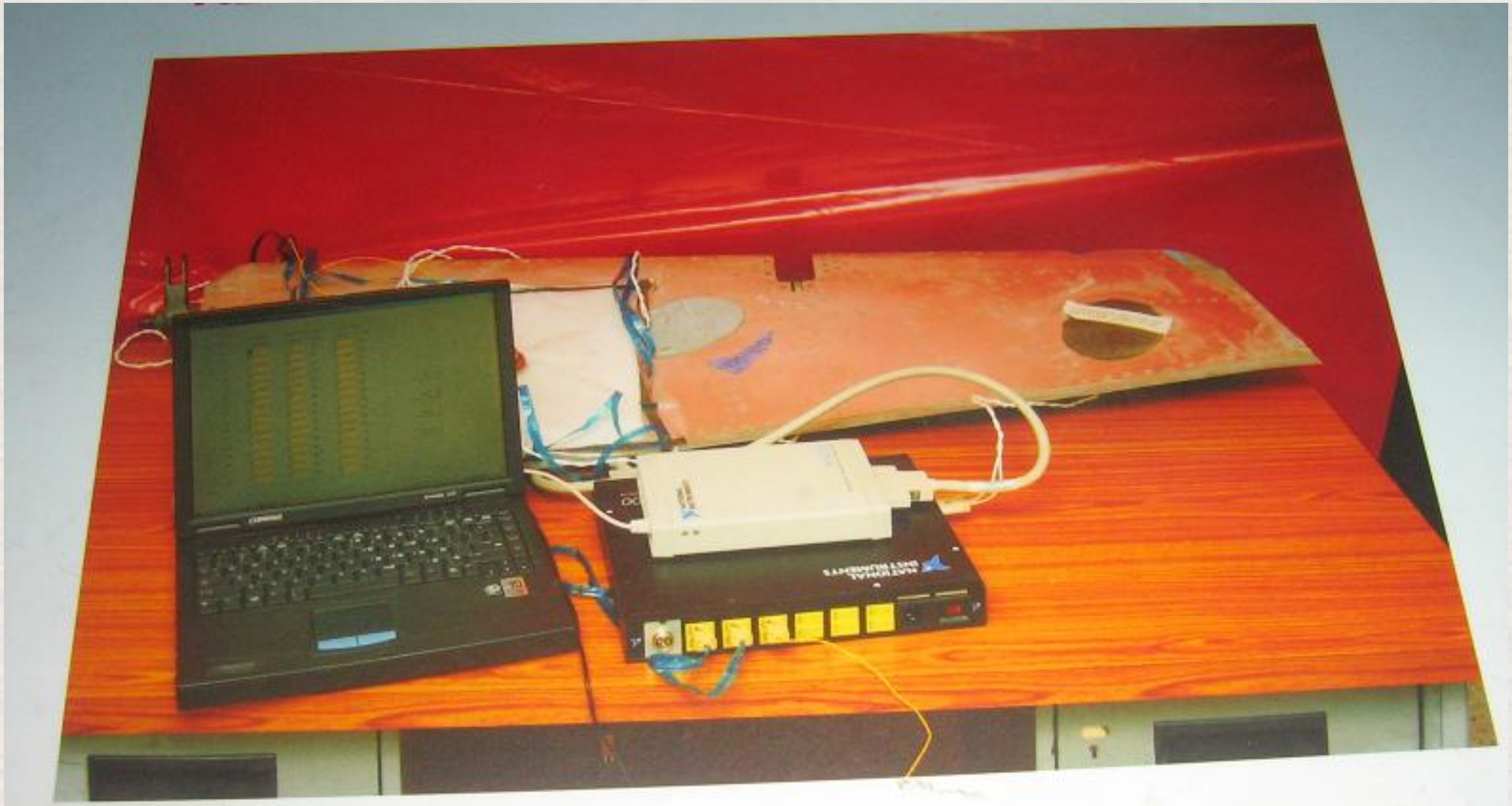
- **REPAIR VALIDATION :** By NDT

CURE-IN PLACE REPAIR



Composite Repair on MiG-21 Rudder

CURE IN PLACE REPAIR USING CURE CONTROLLER



Composite Repair on MiG-21 Rudder

DEVELOPMENT OF FUEL LEAK REPAIR SCHEMES FOR SEA HARRIER

INTEGRAL FUEL TANK REPAIR

- **TYPE OF DAMAGE :**

  FUEL LEAK THROUGH RIVETS AND JOINTS
- **CAUSE OF DAMAGE :**

  AGEING OF SEALENTS
- **REPAIR REQUIREMENTS :**

  ARREST THE FUEL LEAK
- **REPAIR SCHEME:**

  BONDING OF THIN COMPOSITE LAYER IN SIDE THE FUEL TANK
- **REPAIR VALIDATION :**

  BY TESTING
- **REPAIR QUALIFICATION:**

  BY QA INS HANSA & RCMA NASIK

FEATURES OF SEA HARRIER AIRCRAFT FUEL TANK

- **The normal working pressure with full of fuel is 6 lb/sq in (0.41 bar).**
- **There is no barrier between the fuel and fuselage i.e. integral fuel tank.**
- **The integral fuel tanks are constructed with interfying sealant between mating surfaces and around fixings.**
- **This sealant layer is very thin and if structural movement occurs, due to vibration and deflection, the aged sealant over-strains and fails.**
- **Once the interfaying sealant fails due to the ageing, there is no access to re-apply the sealant, because the sub structures are connected with plenty of rivets.**
- **In the current repair scheme, the application of sealant is only on the surface instead of interfying, which is not sufficient to arrest the leakage of fuel on a long run and there fore frequent repair is needed.**
- To overcome the problem, a **Repair Scheme of Laying Thin Composite Skin Inside the Fuel Tank** is suggested as a permanent solution.

CENTRE FUEL TANK OF SEA HARRIER AIRCRAFT

FRONT FUEL TANK OF SEA HARRIER AIRCRAFT

CENTRE FUEL TANK OF SEA HARRIER AIRCRAFT

CENTRE FUEL TANK OF SEA HARRIER AIRCRAFT

CENTRE FUEL TANK OF SEA HARRIER AIR CRAFT

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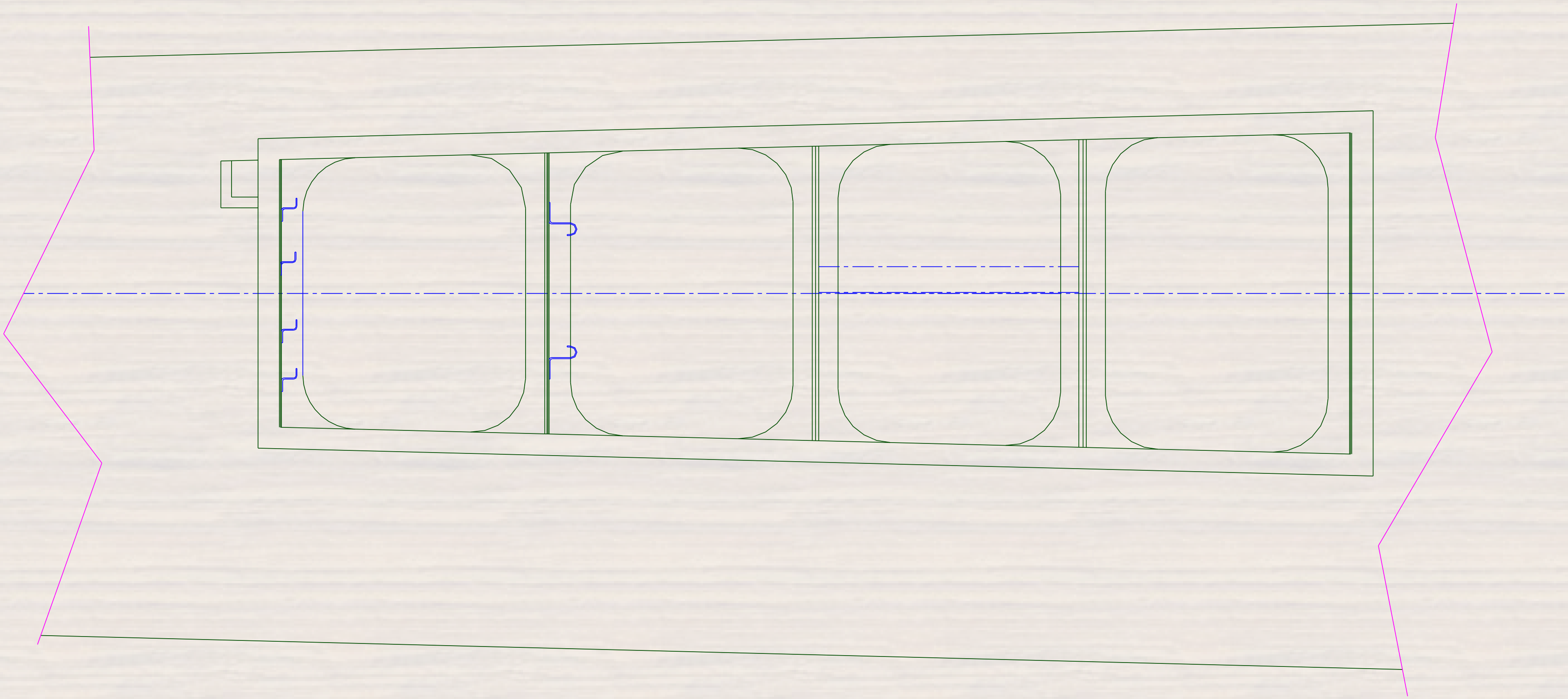
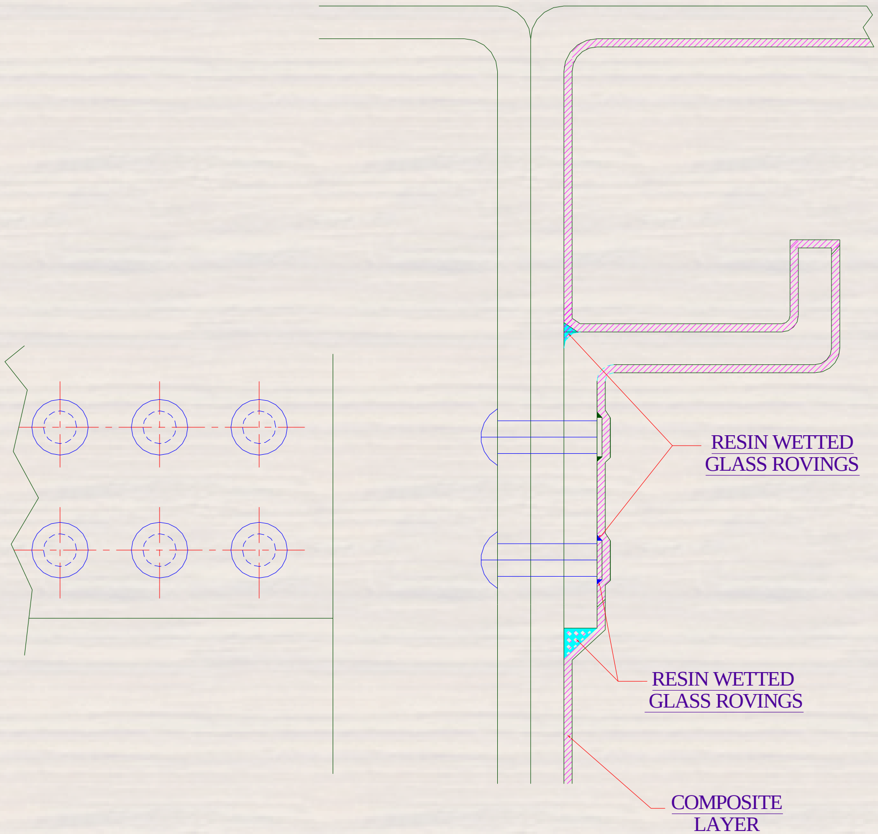
CENTRE FUEL TANK OF SEA HARRIER AIRCRAFT

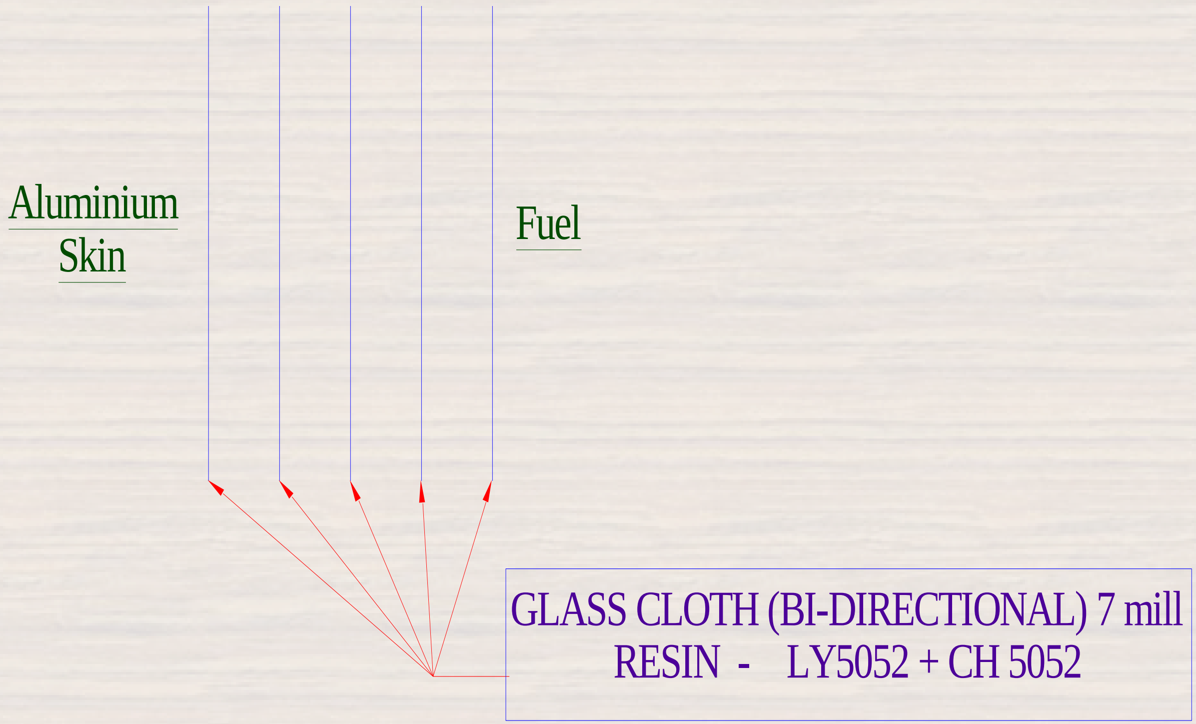


REPAIR SCHEME OF CENTRE FUEL TANK OF SEA HARRIER AIRCRAFT



LAYING THE COMPOSITE SKIN

REPAIR SCHEME OF CENTRE FUEL TANK OF SEA HARRIER AIRCRAFT



LAYER DETAILS

REPAIR VALIDATION

REPAIR PROCESS

DESIGN APPROVAL

PROCESS APPROVAL BY RCMA

REPAIR IMPLIMENTATION ON AIRCRAFT

REPAIR QUALIFICATION BY TESTING

REPAIR SCHEMES FOR LIGHT COMBAT AIRCRAFT (LCA) TEJAS

SKIN DAMAGE

Cause of damage :

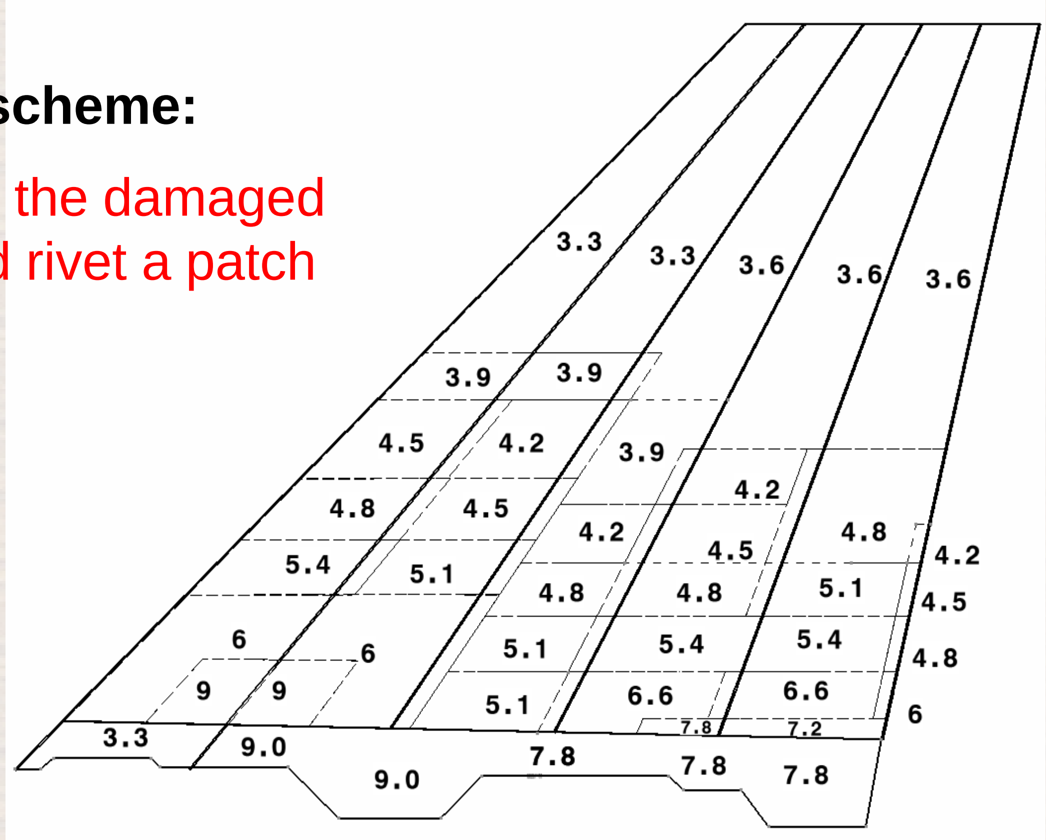
Impact

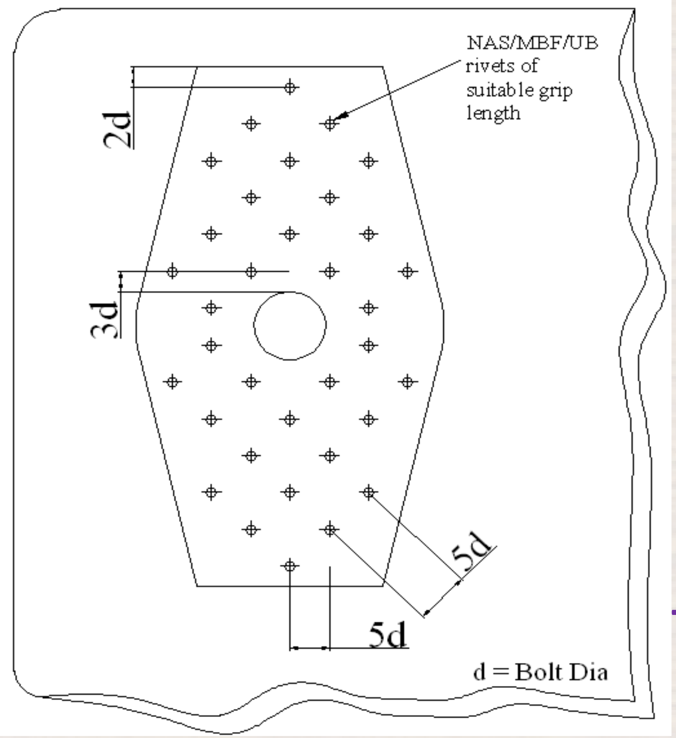
Type of damage :

Skin puncture at t ≥ 6mm

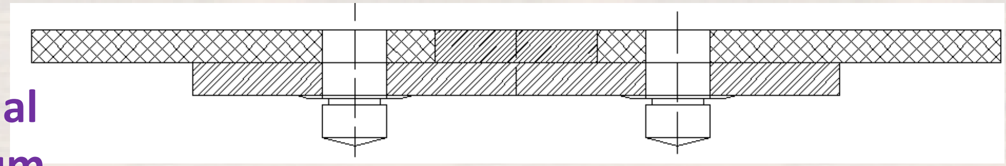
Repair scheme:

Remove the damaged area and rivet a patch





External Titanium patch



Internal riveted patch

LCA-EXTERNAL METALLIC PATCH REPAIR

TYPE OF DAMAGE: Crack in wing spar of LCA

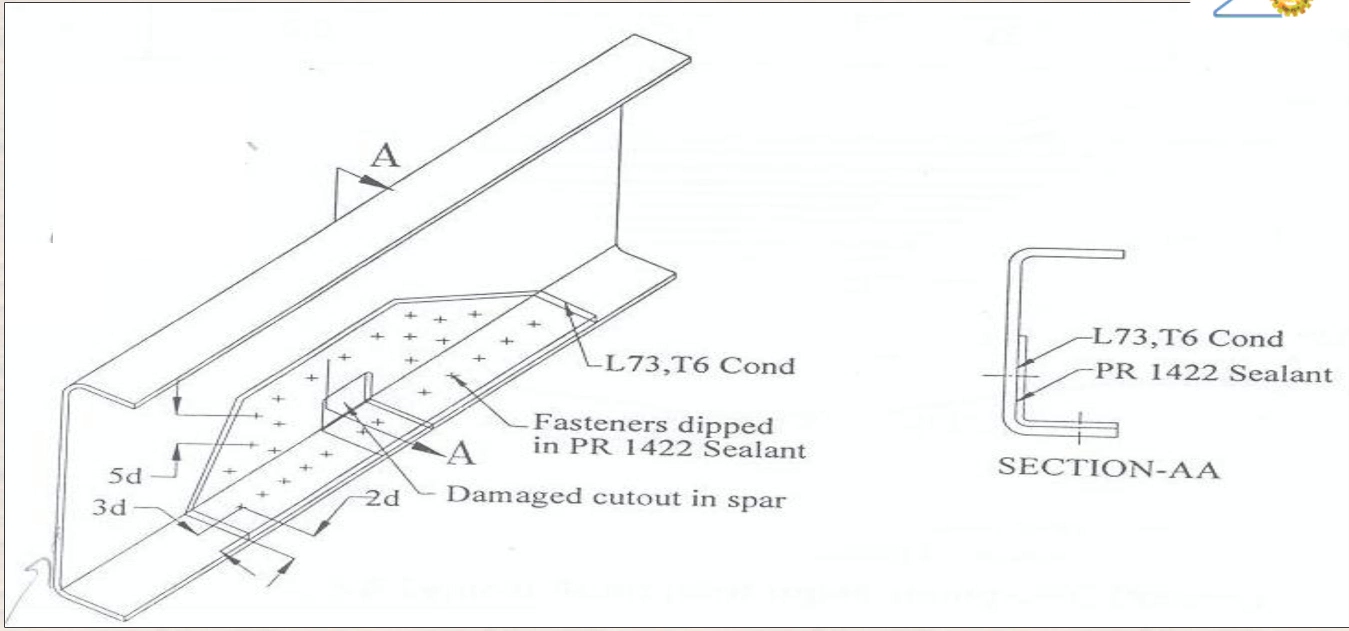
CAUSE OF DAMAGE : FOD

REPAIR REQUIREMENTS:

Restore Structural integrity & Load transfer

REPAIR SCHEME:

External patch repair



Metallic Patch selection:

Material : Shear load transfer (**S_P**)

No. of fasteners (N) : Shear allowable load or Laminate bearing stress whichever is critical

Fastener location : 3d edge distance and 5d spacing between fasteners

Fastener type : MBF-3003-5-250/NAS-1919C-05S-05

Shear load to be transferred, $S_P = б_{xy}\, lt$

$N_{SHEAR} = P_P / V_{all}$

$N_{BEARING} = P_P / F_{BRU}\, d\, t_s$

$б_{xy}$ –Equivalent shear strength of the laminate

P_P – Axial load in patch

l – Length of the joint

t / t_s – Thickness of the laminate

d – Hole/ Fastener diameter

V_{all} – Single shear allowable load of fastener

F_{BRU} – Allowable bearing strength of laminate

DESIGN AND DEVELOPMENT OF A REPAIR SCHEME FOR CRACKED MAIN LANDING BEAM (PORT SIDE) OF MIG-23 UB MS-315 AIRCRAFT

TYPE OF DAMAGE : Crack in MLG beam

CAUSE OF DAMAGE : Stress corrosion

REPAIR REQUIREMENTS :

- Restoration of structural integrity

REPAIR SCHEME:

- Scarf joint repair & hybrid joint

REPAIR VALIDATION : By NDT & testing

Repair of MLG beam of MiG-23 Aircraft

MLG Beam of MiG-23

MLG beam in extended position

MLG in retracted condition

INTRODUCTION

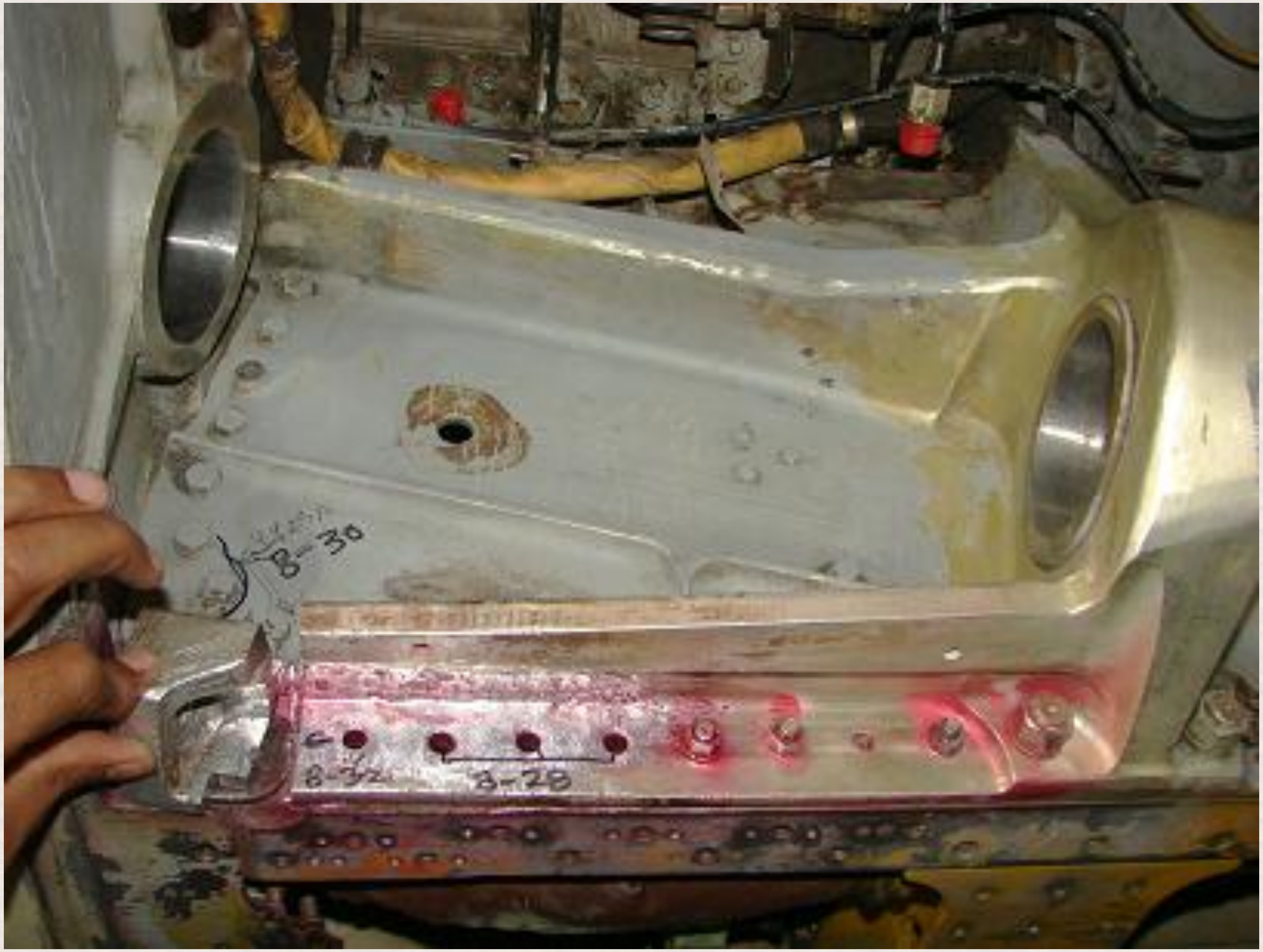
- MS-315 Main Landing Gear Beam cracked due to Stress corrosion cracking (material inherent property – induced residual stresses)
- 80 % OF MiG-23 UB (trainers) are detected with MLG beam cracks – residual stress induced

➢ 2 repair schemes were existing from OEM for 66 mm crack initially and upto 180 mm crack subsequently

➢ First 66 mm portion of the beam being cut out by OEM since origin of crack is within the material.

➢ Crack length in this case is of 210 mm and is 30 mm more than OEM permitted value. Hence No Conventional repair will work. So 11BRD approached NAL .

CRACKED MLG BEAM



REPAIR SCHEME

CFRP Patch Design Considerations:

- Design of CFRP Repair patch such that it restores strength and the crack will not play a major role, as in load transfer is through the patch
- Select a suitable adhesive with fairly high shear strength(VK-9 has 20 MPa).
- Integrate the cut with bolt attachment point at Frame #20
- Composite patch should not interfere with under carriage fairing

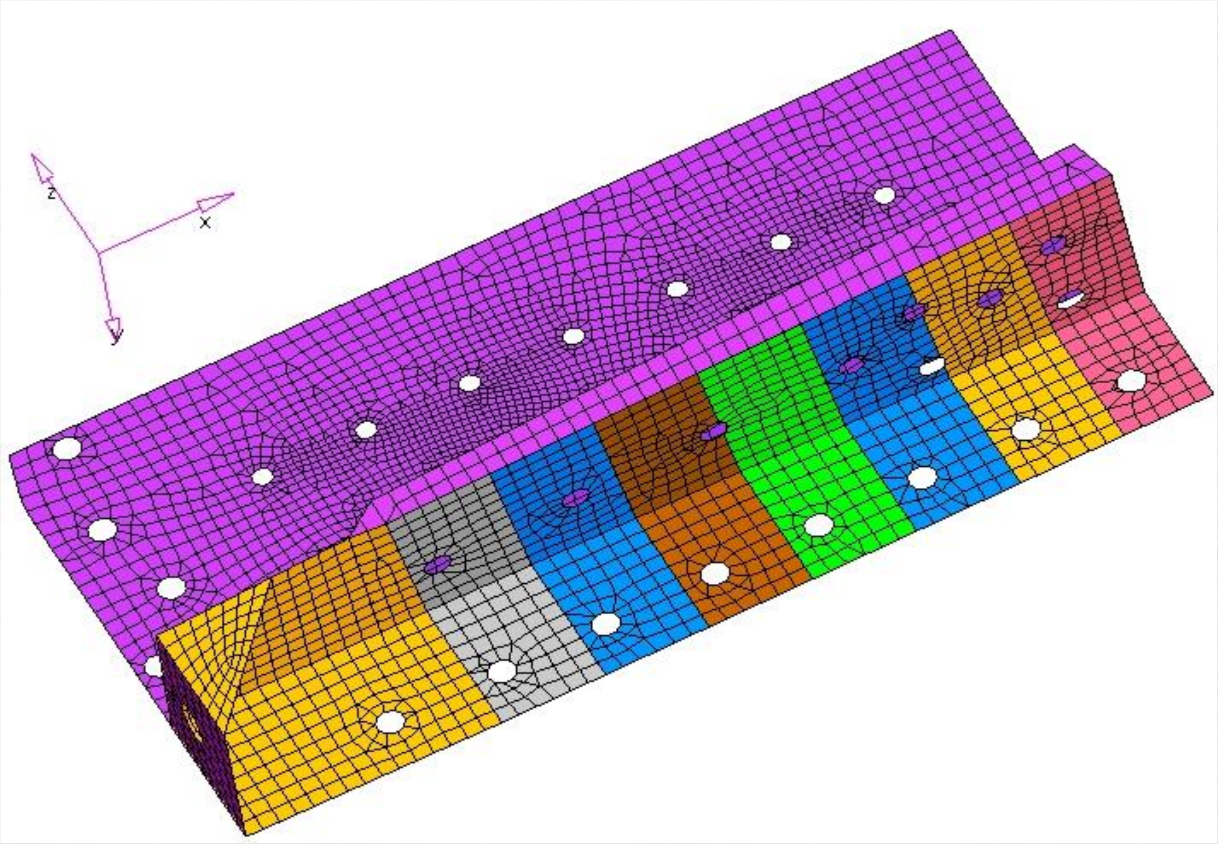
DESIGN PHILOSOPHY

- From the stress album of MiG-27 decide on patch thickness and size
- Fabricate patch by autoclave moulding
- Based on patch thickness select grip length and Bolts
- Patch installation by Hybrid Bonding technique

Repair Design based on:

a. Patch Material
b. Patch Dimension
c. Structural Adhesive

ANALYSIS OF MLG BEAM WITH COMPOSITE PATCH-FE MODEL



Pre &Post Processing-
Hyper mesh

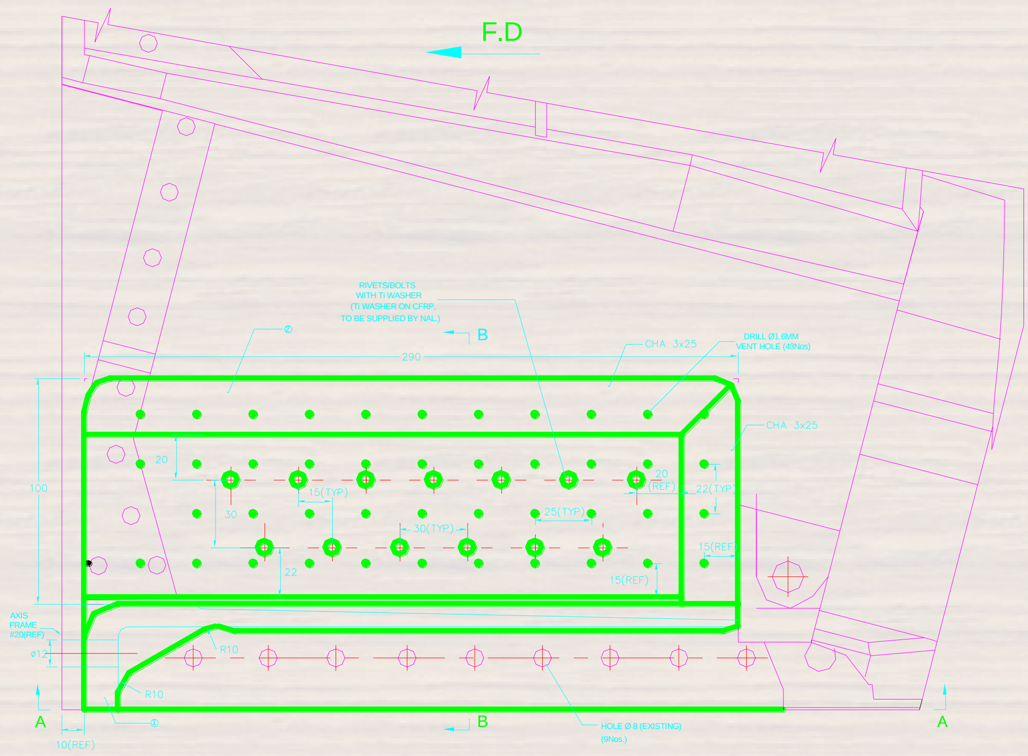
Solver
MSC/Nastran
Patch modelling
2D layered shell elements,
(QUAD 4 & TRIA 3) with PCOMP properties

Fasteners to composite patch

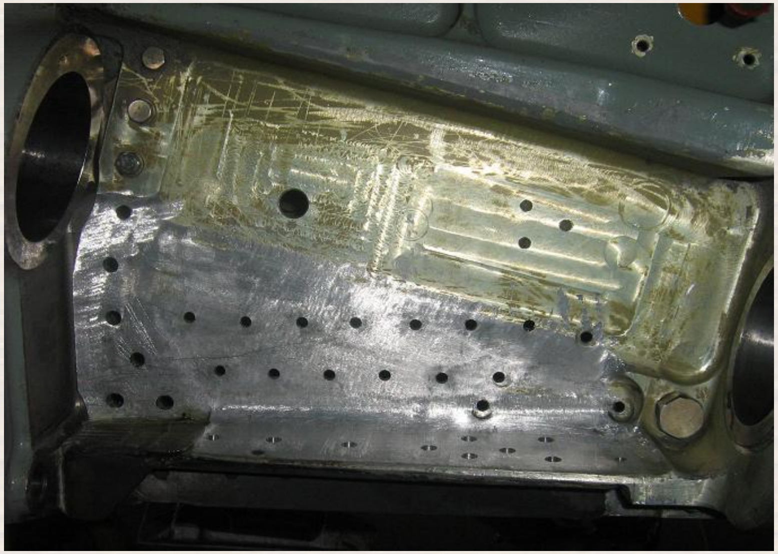
By RBE-2 elements
Bonding not modeled

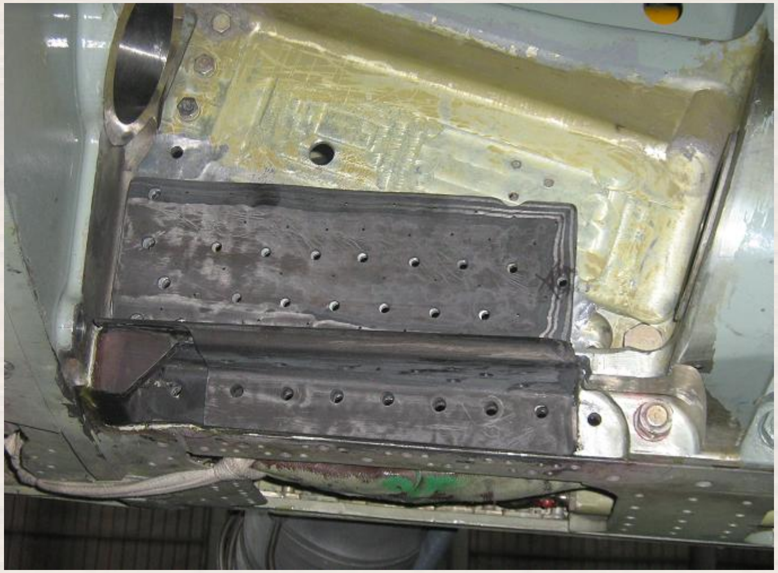
- Composite repair scheme modeled and load transfer ensured

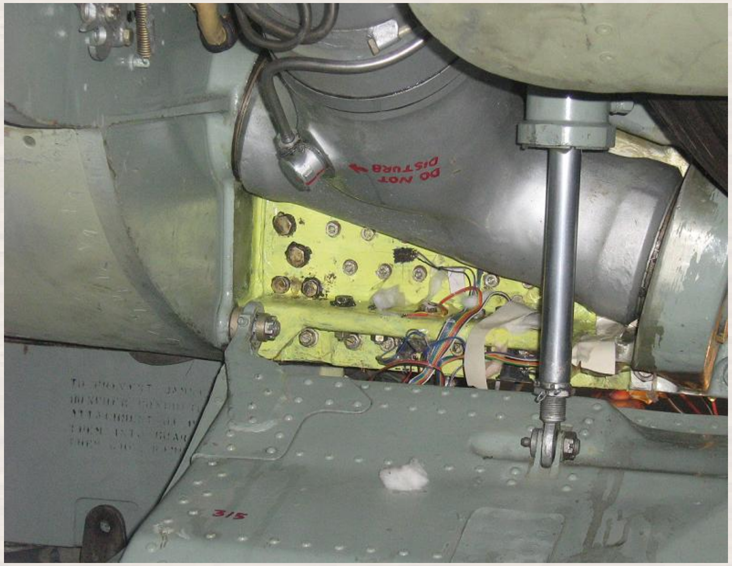
CFRP PATCH INSTALLATION



REPAIR SEQUENCE

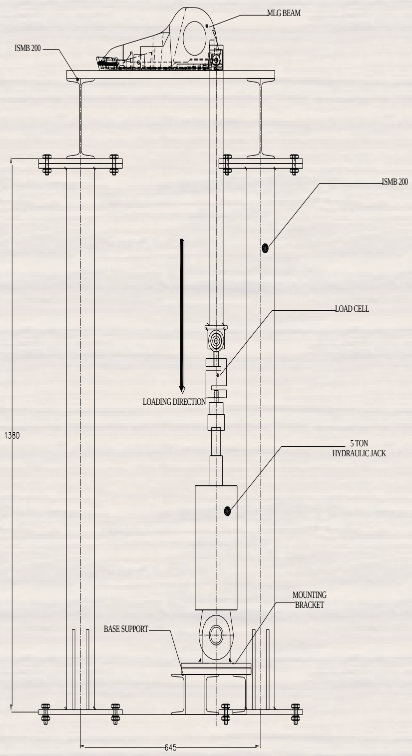


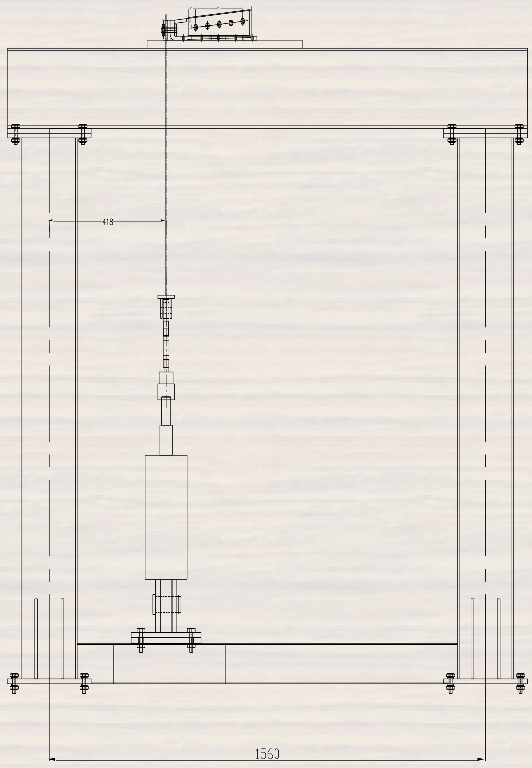




MLG BEAM SURFACE PREPARATION FOR ADHESIVE BONDING

STRUCTURAL TESTING
ANGULAR FITTING JOINT AT FR #20

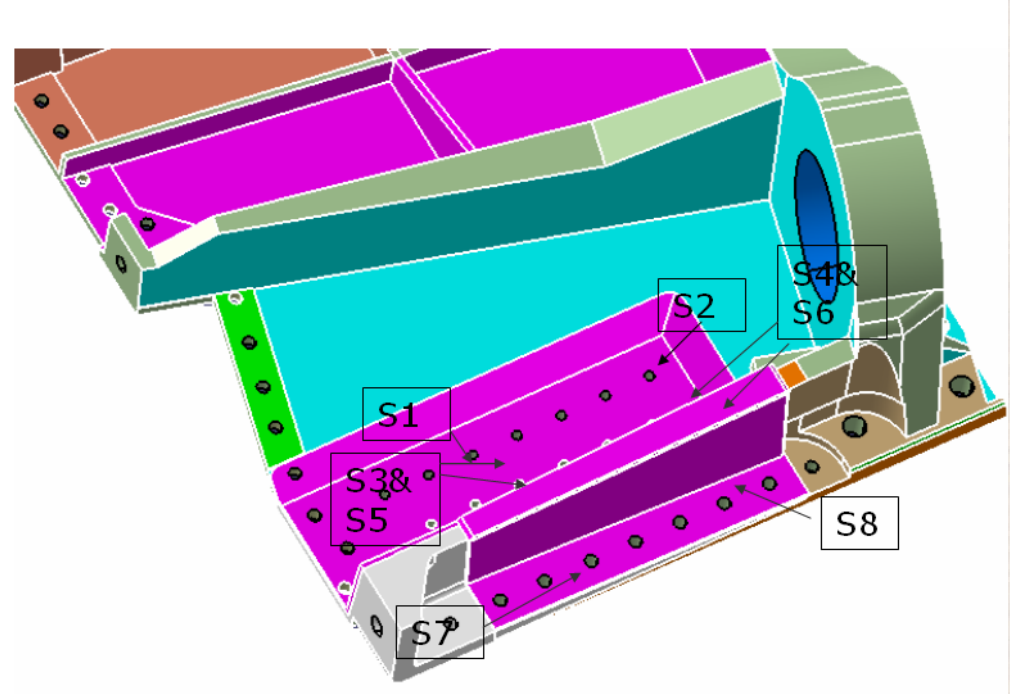




* Load = 2000 Kg as per stress album

* No local bearing failure at joint

STRUCTURAL TESTING ON AIRCRAFT
STRAIN MEASUREMENT ON COMPOSITE REPAIR



Strain gauge locations

on Repaired MLG beam

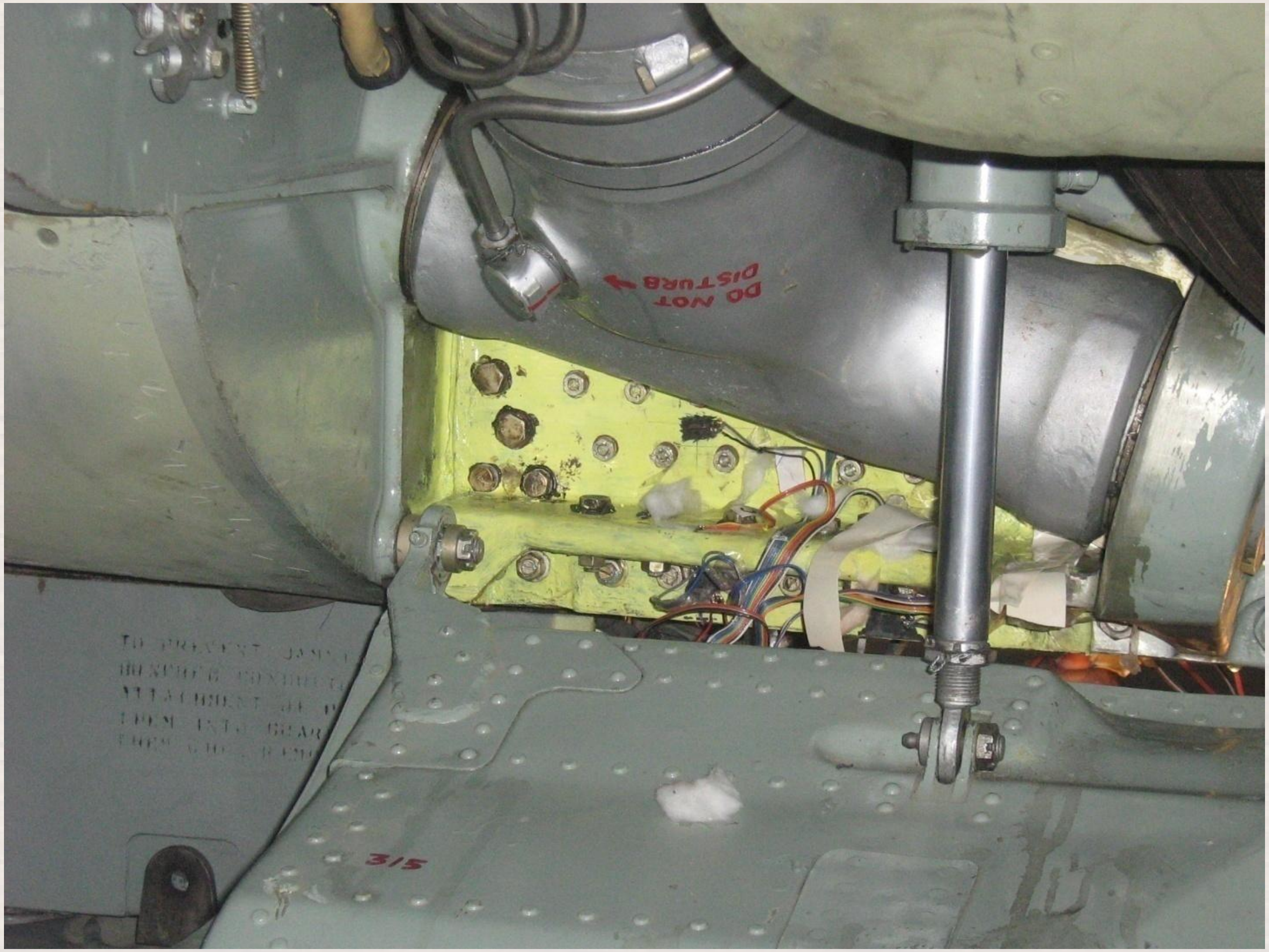
Strain Gauge No	Strain (Micro Strains)
S1	11
S2	35
S3	96
S4	140
S5	91
S6	**186**
S7	-26
S8	-38

- All gauges came back to initial zero condition

Structural Testing on Aircraft (MS-308)
Strain Measurement on Good Aircraft

Strain Gauge No	Strain (Micro Strains)
S1	-51
S2	-78
S3	-32
S4	14
S5	-61
S6	06
S7	-134
S8	-76

MLG BEAM AFTER COMPOSITE REPAIR

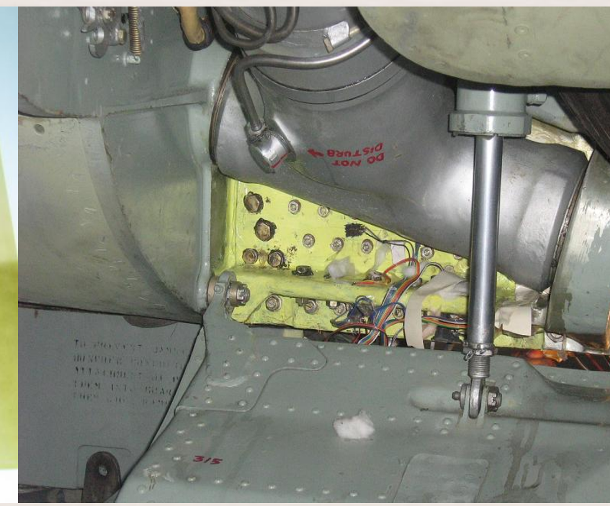


SUMMARY

- It is found that from design that it is possible to restore structural strength of the cracked MLG Beam with composite patches. The CFRP angular patch and cap patches have been attached to the parent MLG Beam by hybrid fastening technique.

DESIGN OF COMPOSITE STRUCTURAL JOINTS





REPAIR OPTION

- Factors to be considered for the selection of a repair joint :

  1) Laminate thickness

  2) Laminate size

  3) Joint strength

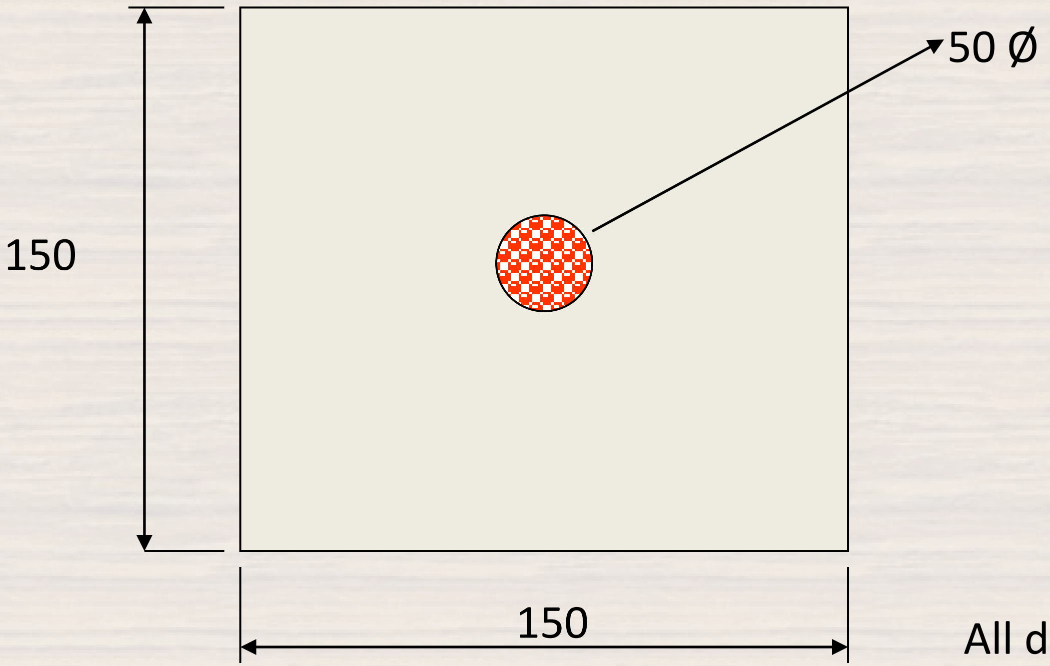
- Before repair design, selection of the repair joint depends on :

  1) Shear stress distribution in the overlap region

  2) Failure of the joint

PROBLEM DEFINITION

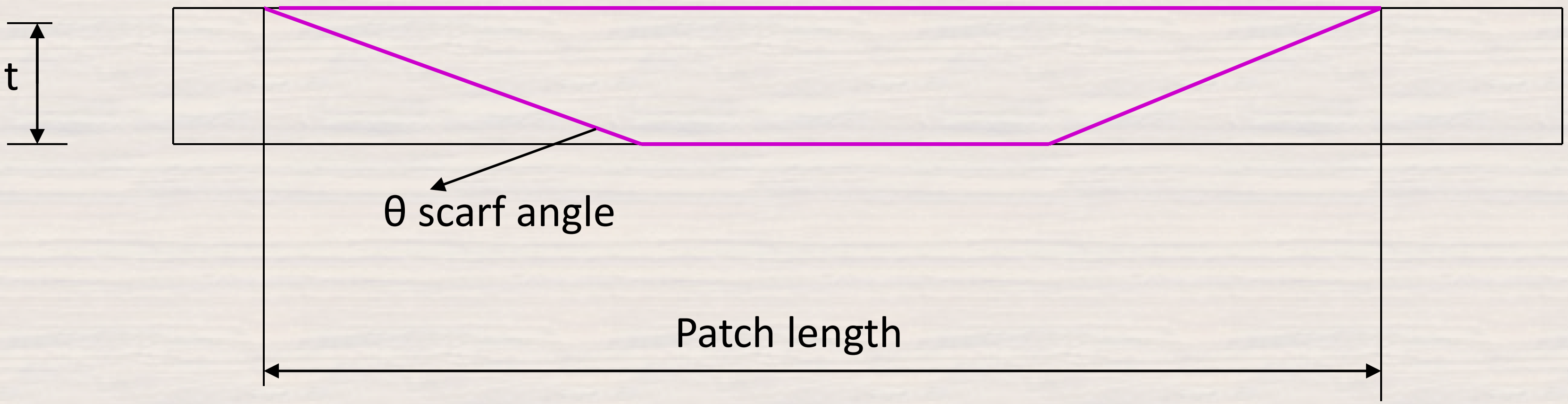
- Develop repair schemes for a CFRP laminate of size (150 x 150 x 6)mm with an impact damage of size 50mm dia.



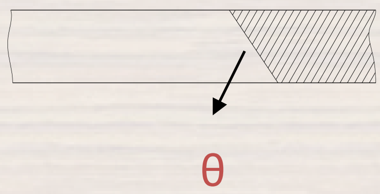
All dimensions in mm

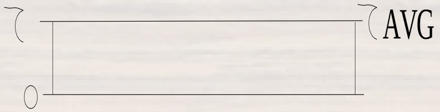
REPAIR OPTION 1- SCARF JOINT

- As the thickness is more than 3 mm, scarf joint is one of the options
- Further in a scarf joint , Shear stress in the adhesive layer is uniform
- For taking the full advantage of Scarf joint, provide a taper of 1:20



SCARF DESIGN





As the shear stress distribution in a Scarf joint is constant, simple theory gives

✓ $\zeta = P \sin 2\theta / 2t$ and $\sigma = P \sin^2\theta / t$

ζ = shear stress, σ = normal stress, θ= Scarf angle

- Min value of scarf angle, for applied load is,

✓ $P = E\, e_u\, t$

$= 2\, \zeta_p\, t / \sin 2\theta$

- Thus for smaller θ, condition for allowable strain is

$$\theta < \zeta p / E\ eu$$

Design allowable strain e_u = 4000 micro strain,
ζp = 20 MPa,
E= 72 GPa

Thus, $\theta \leq 3^{\circ}$

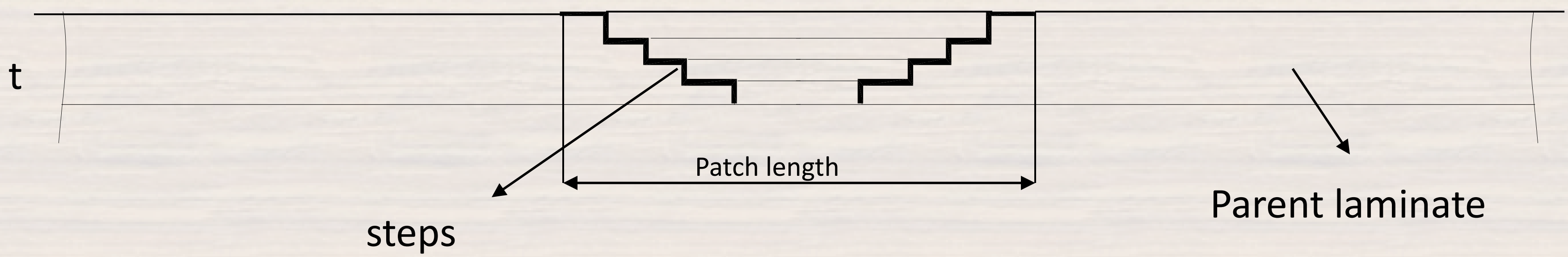
- Taper length, L= t/ tan θ (taper, 1:20)
- Total patch length= (2 * taper length) + damage size
- Therefore, only **damage dia. ≤ 25 mm** is possible in a laminate size of (150x150x6)mm to repair by a scarf joint.

SCARF DESIGN LIMITATIONS

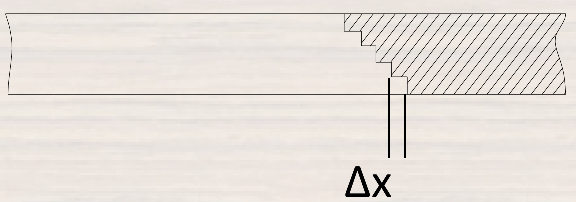
- Damage size upto 25 mm dia for a (150 X 150 X 6) mm laminate
- Thickness limitation and skilled labor
- If the patch and the parent material varies in stiffness and thickness, the shear stress is no longer a constant.

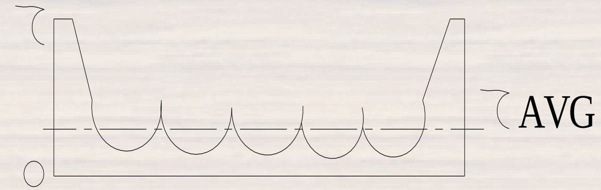
REPAIR OPTION 2 : STEPPED LAP JOINT

- ✓ It has a non- uniform shear stress distribution with high stresses at the end of each step
- ✓ Load carrying capacity can be increased by increasing the number of steps



STEPPED LAP JOINT DESIGN





- Step Length = Δx

- $\zeta = \Delta P / \Delta x$

- If it is assumed that load increment ΔP on each step is proportional to the relative stiffness of the ply, ΔP / P = stiffness of ply / total stiffness

- Approx. required step length Δx can be made by assuming the adhesive is stressed to its shear yield stress ζp and each of the plies loaded individually,

$$\Delta x = (E_P \quad e_u / \zeta_p). \text{ ply thickness}$$

- Where E_P = ply stiffness = 130 GPa for 0° plies

  e_u = 4000 micro strains

  Thus, Step length, Δx= 3.9 mm

- +/- 45 and 90 plies are also made to same length

- Therefore,
  6mm thick laminate = 40 plies @ 1 ply thickness= 0.15mm

  1 step = 10 plies

  No. of steps = 4

  Lap length = (4 x 3.9)mm = 15.6 mm

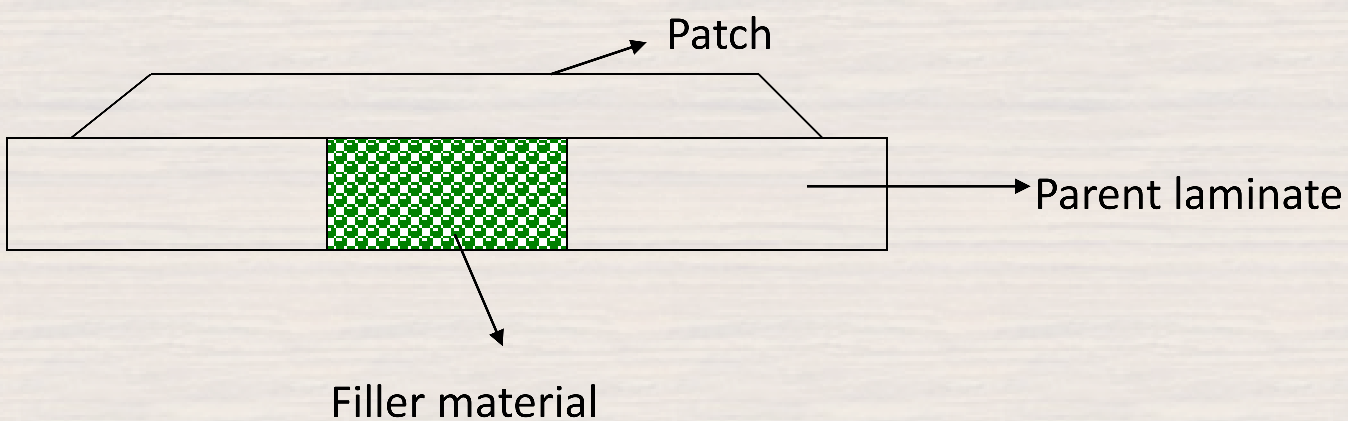
  Total patch length= (2 x lap length) + damage size

  For a damage size of 50 mm dia is total patch length is 81.2mm.

  Hence, **damage dia. of 50 mm** is possible in a laminate size of (150x150x6)mm to repair by a stepped lap joint.

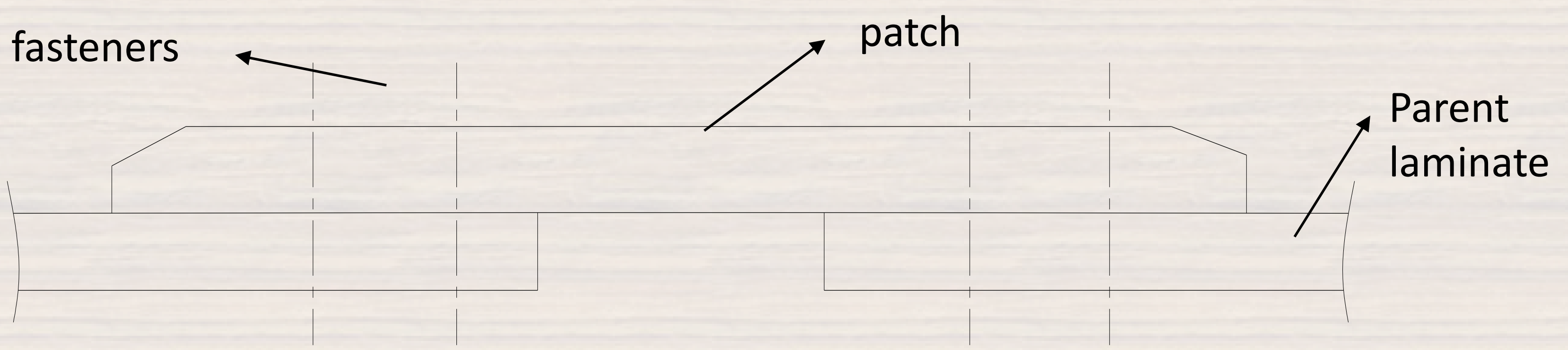
REPAIR OPTION 3 : BONDED EXTERNAL PATCH JOINT

- Preferable for thick laminates
- Easily applied under field conditions
- Outer taper is required to minimize peel stresses at the ends of the patch



REPAIR OPTION 4: BOLTED EXTERNAL JOINT

- Preferable for thick laminates
- They are employed in field repairs
- Titanium alloy is employed for metal patch, nut plate fasteners, since it does not suffer galvanic corrosion



- Produce stress concentration
- Susceptible to delamination from drilling/machining

SUMMARY

- Damage dia. ≤ 25 mm can be repaired by scarf joint for a (150x150x6)mm laminate
- Stepped lap joint design is the repair option suitable to repair a damage size of 50 mm dia. in a (150 x 150 x 6) mm laminate

CONCLUDING REMARKS

1. Conventional composite repair techniques use structural grade prepregs and adhesives under vacuum and high temperature curing
2. Bonded repairs preferred over Bolted repairs. Generally in bonded repairs one has to design the patch to match the strength and stiffness of the material
3. Hybrid joints give better static as well as fatigue performance than any conventional joint configurations, i.e., bonded or hybrid joints.
4. To provide a fail-safe design, use hybrid joints(bonded/bolted) in composites structures
5. Advanced Composites are preferred to repair cracked metallic structures due to their inherent directional properties and ability to confirm complex contours, without initiation of self simulated cracks like metallic patches
6. Further Composite repair can be used either in the Field or in Depot level repairs
7. **Selection of a repair joint for an aircraft structure depends on** Shear stress distribution in the overlap region, load transfer through the joint, Failure of the joint, repair time and weight
8. **Current trends in aircraft repair are use of advanced composites to repair metallic & composite aircraft structures and development of machining tools for speedy repair**

REFERENCES

(1) AA BAKER "Repair of cracked or Defective metallic Aircraft components with Advance fiber composites an overview of Australian work" Composite structures 2 (1984)153-181.

(2) Keith B Armstrong, Richard T Barrett "care and repair of Advanced composites" SAE International, 1998.

(3) Michael CY Niu "Composite Airframe structures Practical Design Information Data"-CONMILIT press LTD, Hong Kong, 1992.

(4) Hoskin BC and Baker AA "Composite Materials for Aircraft Structures" ,AIAA Educational Series 1986

(5) Hart-Smith LJ "Analysis and Design of Advanced Composite Joints", NASA-CR-2218, 1974

(6) Sqn Ldr SH Chicken and Wg Cdr Welburn" Repairing composite structures-RAF experience of peace Time and Battle Damage Techniques"-The Aeronautical Journal 1997

(7) AGARD Conference Proceedings-550 "Composite Repair of military Aircraft Structures ", 1995.

(8) Proceedings of "Workshop on repair technology for composite structures" vol-1 (theory) NAL Doc.No.PR-AC-01-01, 2001

(9) AA BAKER "Advances in the bonded composite repair of metallic aircraft structure" Volume 1& 2- Elsevier 2002

(10) H. Qian and C.T. Sun, "Performance of a composite double strap joint with attachments," Joining and Repair of Composite Structures ASTM STP 1455 (2004)

(11) C.T. Sun et.al. "Development of Improved Hybrid Joints for Composite Structures" Purdue University,2010

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